

NASA Conference Publication 10003

Aeropropulsion '87

Session 6— High-Speed Propulsion Technology

Preprint for a conference at
NASA Lewis Research Center
Cleveland, Ohio, November 17-19, 1987



National Aeronautics and
Space Administration

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NASA THRUSTS IN HIGH-SPEED AEROPROPULSION RESEARCH AND DEVELOPMENT - AN OVERVIEW

Joseph A. Ziemianski

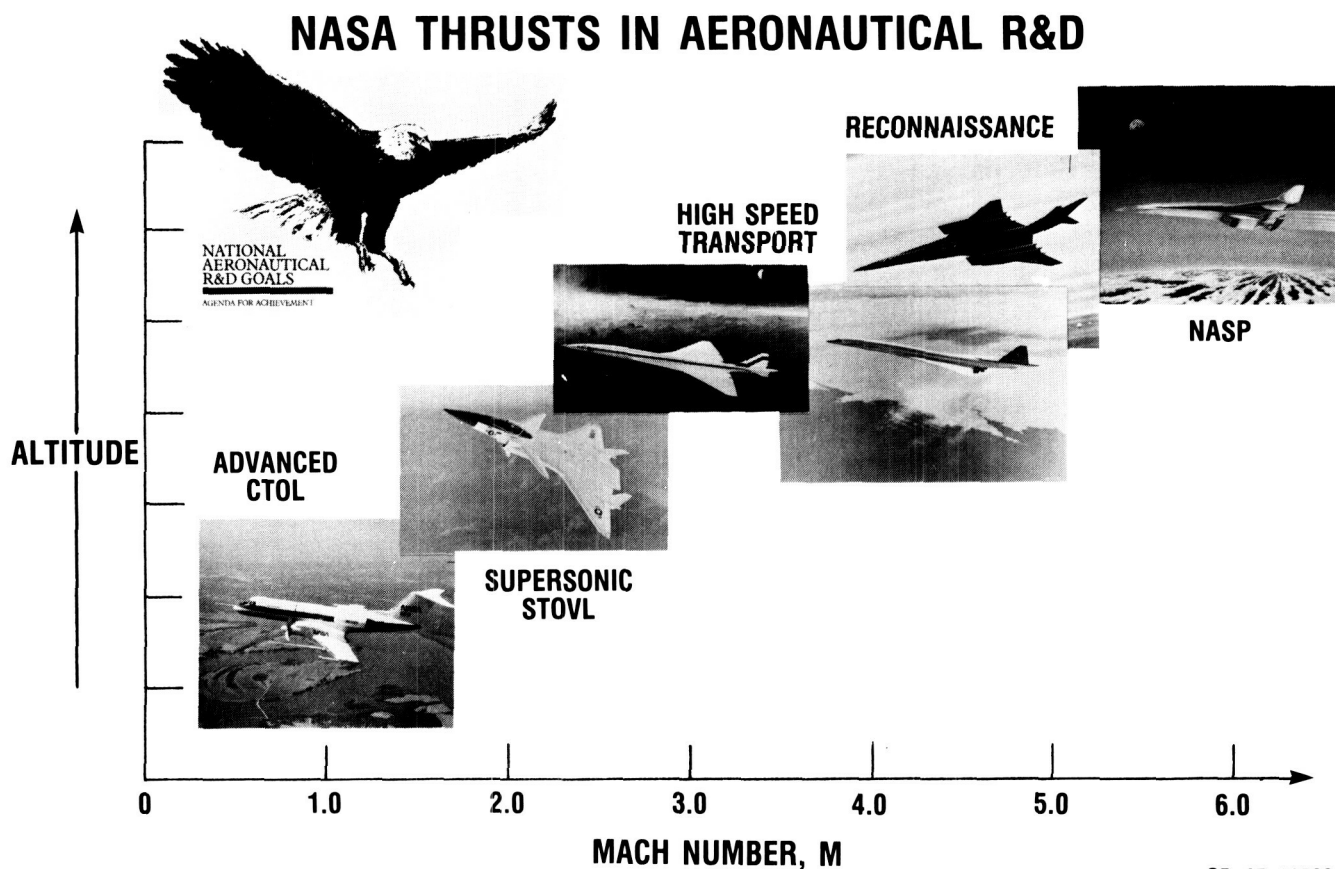
ABSTRACT

NASA is conducting aeronautical research over a broad range of Mach numbers. In addition to the Advanced CTOL propulsion research described in a separate session at this conference, the Lewis Research Center has intensified its efforts towards propulsion technology for selected high-speed flight applications. In a companion program, the Langley Research Center has also accomplished excellent research in Supersonic Combustion Ramjet (SCRAM) propulsion. What will be presented in this session is an unclassified review of some of the propulsion research results that are applicable for supersonic to hypersonic vehicles. This overview not only provides a preview of the more detailed presentations which follow, it also presents a viewpoint on future research directions by calling attention to the unique cycles, components, and facilities involved in this rapidly expanding area of work.

NASA THRUSTS IN AERONAUTICAL RESEARCH AND DEVELOPMENT

The Lewis Research Center is conducting high-speed propulsion research over a broad range of Mach numbers. In addition the Langley Research Center has accomplished related research in Supersonic Combustion Ramjet (SCRAM) propulsion. What will be presented here is an overview of some of the propulsion research results that are applicable to the classes of supersonic/hypersonic vehicles shown on the figure. The research program associated with the National Aerospace Plane (NASP) will not be discussed because of classification, but the Langley Research Center will present a paper on Hypersonic Propulsion Research.

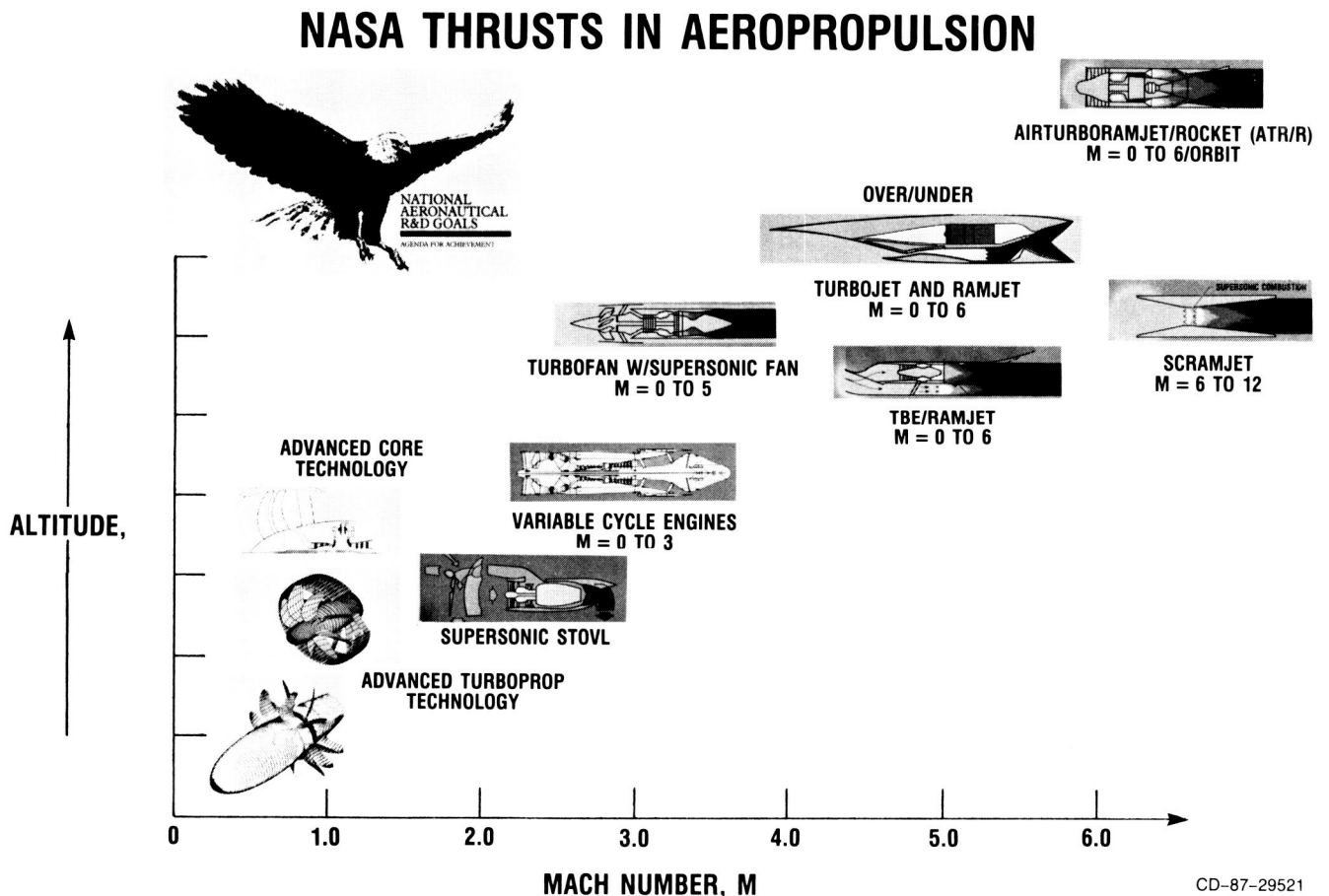
A separate session at this conference described the Advanced CTOL propulsion research results.



NASA THRUSTS IN AEROPROPULSION

The propulsion concepts that NASA has identified with the vehicle classes shown previously are illustrated in this figure. What will be presented in this High-Speed Propulsion Technology Session will be as follows:

- (1) An evaluation of the merits of a High-Speed Transport and the associated propulsion technology barriers
- (2) A further exposition of a propulsion research concept that may have applicability to the High-Speed Transport, a Supersonic Through Flow Fan
- (3) High-speed inlet studies in the Mach 5 range
- (4) Propulsion research results associated with Supersonic Short Takeoff and Vertical Landing (SSTOVL) vehicle
- (5) A summary of hypersonic propulsion research at the Langley Research Center



FUTURE DIRECTIONS IN HIGH-SPEED PROPULSION RESEARCH AND DEVELOPMENT

The future of NASA's propulsion research will shift to the higher Mach propulsion concepts because of the renewed emphasis on such challenging programs as SSTOVL, High-Speed Transports, and NASP. What is required is the identification of propulsion unique cycles and associated components that could lead to new vehicle capabilities in the high Mach regimes. Because of the complexity and limited data base at high Mach numbers, considerable discipline research must first be accomplished before any component program can begin. Integration of the propulsion system is complex; because of the high Mach numbers the propulsion and airframe become an integral system.

FUTURE DIRECTIONS IN HIGH-SPEED PROPULSION R&D

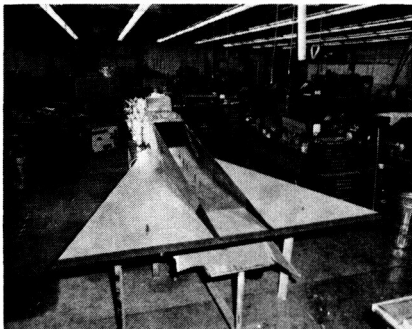
- **MAJOR EXPANSION OF NASA THRUSTS IN HIGH-SPEED PROPULSION TECHNOLOGIES**
- **UNIQUE CYCLES AND COMPONENTS BEING ADDRESSED THAT COULD LEAD TO NEW VEHICLE CAPABILITIES IN SUPERSONIC/HYPERSONIC FLIGHT REGIMES**
 - DISCIPLINE RESEARCH**
 - COMPONENT/SYSTEM INTEGRATION**

UNIQUE TECHNOLOGIES FOR HIGH-SPEED PROPULSION

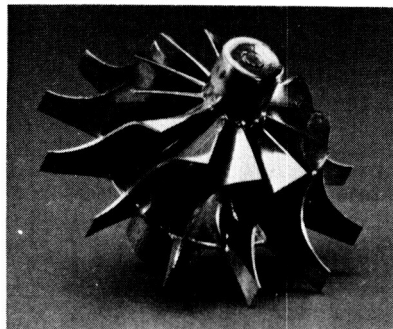
Unique technologies required for high-speed propulsion are shown on this figure. They range from high-speed inlets and the associated computational fluid dynamic codes needed to address the complex flows to high temperature materials/structures needed in the hot section of the engines. For application such as the High Speed Civil Transport, unique nozzles and suppressors are required for performance and community noise considerations.

Supersonic compression is an identified technology that can lead to more fuel efficient propulsion concepts, in the Mach 2 to 4 range. Supersonic combustion is needed around Mach 6 for a scram propulsion mode but may have promise at lower speeds in a complete supersonic through flow machine. How all these technologies are brought together in a viable propulsion concept is the challenge of today's researchers.

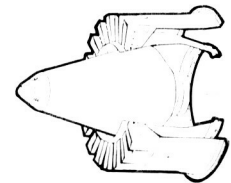
UNIQUE TECHNOLOGIES FOR HIGH-SPEED PROPULSION



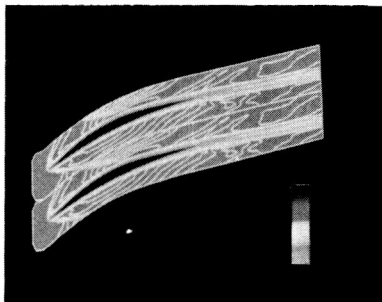
HIGH-SPEED INLETS



**HIGH TEMPERATURE
MATERIALS/STRUCTURES**



**ADVANCED NOZZLES
AND SUPPRESSORS**



SUPERSONIC COMPRESSION



SUPERSONIC COMBUSTION

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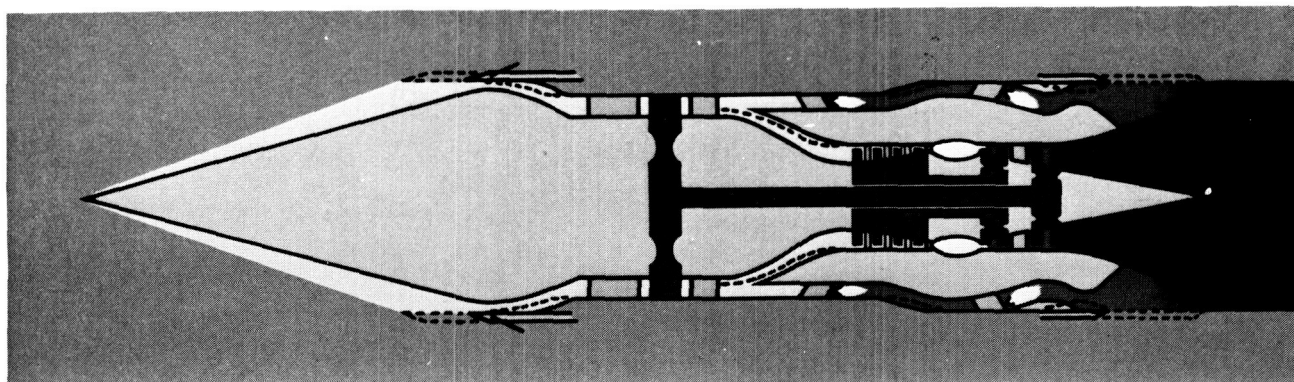
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PUTTING IT ALL TOGETHER

One configuration that shows considerable promise is the Supersonic Through Flow Engine. but as previously described it will require major progress in a variety of discipline and component research areas and careful matching of those elements. If successful, our analyses indicate for a High Speed Civil application that this concept will be 10 percent more efficient and the weight will be 25 percent less than that of a variable cycle engine.

PUTTING IT ALL TOGETHER

A SUCCESSFUL NEW ENGINE WILL REQUIRE NOT ONLY MAJOR PROGRESS IN DISCIPLINE AND COMPONENT RESEARCH, BUT ALSO THE CAREFUL MATCHING AND COORDINATION OF THOSE ELEMENTS.



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HIGH-SPEED AEROPROPULSION RESEARCH WIND TUNNEL FACILITIES

NASA Lewis Research Center has positioned itself in the high-speed regime by developing additional capabilities in our complex of testing facilities. We have added in our Propulsion Systems Laboratory (PSL) the capability to run turbomachinery to Mach 6 with gaseous hydrogen and oxygen; and in our 1 x 1 foot tunnel heaters have been added to eliminate condensation shocks to Mach 6. This couples with the current 10 x 10 supersonic (Mach 3.5) and 8 x 6 transonic tunnels to give NASA the capability to test over a broad Mach range.

In addition, we are investigating reopening our Hypersonic Test Facility (HTF) at Plum Brook which if successful will have capabilities to test at Mach numbers of 5, 6 and 7 in a nonvitiated mode. In addition, if heaters are added, the facility capabilities can be expanded substantially.

HIGH-SPEED AEROPROPULSION RESEARCH WIND TUNNEL FACILITIES

- | | |
|---|------------------|
| • 10 x 10 FT S.W.T. | MACH \leq 3.5 |
| • PSL 4 | MACH \leq 6* |
| • 1 x 1 FT TUNNEL | MACH \leq 6* |
| • 8 x 6 FT TUNNEL | TRANSONIC |
| • PLUM BROOK HYPERSONIC TEST FACILITY** | |
| NONVITIATED | MACH 5, 6, AND 7 |
| VITIATED | MACH \leq 6 |

*NEW CAPABILITY

**PROPOSED FOR NASP TESTING

N88-15808

51-07
118889
238

SUPERSONIC STOVL PROPULSION TECHNOLOGY PROGRAM -

AN OVERVIEW

Bernard J. Blaha
and
Peter G. Batterton

ABSTRACT

Planning activities are continuing between NASA, DOD, and two foreign governments to develop the technology and to demonstrate the design capability for advanced, supersonic, short-takeoff and vertical-landing (STOVL) aircraft by the mid-1990's. As a result, a Memorandum of Understanding (MOU) has been established with the United Kingdom to jointly pursue the required technology; and an MOU with Canada is expected to be signed shortly. NASA Lewis Research Center will play a lead role in the development of the required propulsion technologies which have been identified as being "critical" to achieve viable STOVL aircraft. These planning activities have already resulted in initial research programs focused on technologies common to two or more of the proposed propulsion system concepts. This paper will present an overview of the Lewis Research Center's role in the overall program plan and recent results in the development of the required propulsion technologies.

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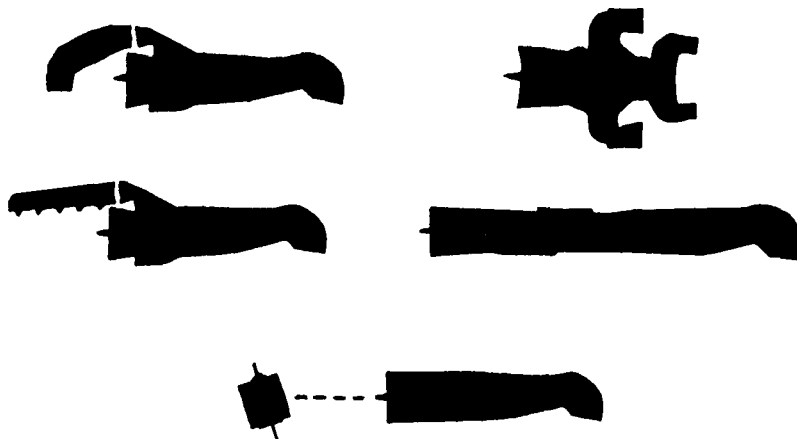
SUPERSONIC STOVL PROPULSION TECHNOLOGY PROGRAM OVERVIEW

This figure shows silhouettes of the five propulsion system concepts being considered for an advanced, supersonic, short-takeoff and vertical-landing (STOVL) aircraft which could be developed in the post-ATF time frame (late 1990's). These include a remote augmenter lift system (RALS), vectored thrust, ejector augmenter, tandem fan, and lift plus lift/cruise systems, respectively. Interest in the development of the technology required for such an aircraft has increased in recent years and has resulted in the initiation of several separate programs and separate Memorandums of Understanding (MOU's) between the U.S. and other governments. An MOU has recently been established with the United Kingdom (U.K.) to jointly pursue the required technology, and an MOU with Canada is expected to be signed shortly. The joint U.S./U.K. program is studying the first four concepts shown. The U.S./Canada program is focused on the ejector concept alone. NASA and the DOD have recently added the lift plus lift/cruise concept to their investigations.

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SUPERSONIC STOVL PROPULSION TECHNOLOGY PROGRAM

OVERVIEW



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PROGRAM GOAL

The supersonic short-takeoff and vertical-landing (STOVL) technology program is focused on having the technologies by the early 1990's, so that a decision to start a research aircraft program can be made with relatively low risk. The key technologies to be developed are primarily propulsion related. We know how to design fighter/attack aircraft which can dash and cruise supersonically (e.g., F-14, F-15, and F-16). We also know how to design a subsonic aircraft for vertical takeoff and landing (e.g., Harrier and AV-8B). The challenge, therefore, is to combine these capabilities into a new, efficient, high performance, supersonic, fighter/attack aircraft for the post Advanced Tactical Fighter (ATF) time frame. This will require the development of unique engine system components with multifunction capabilities (e.g., vectoring and deflecting nozzles and, in particular, new control systems).

PROGRAM GOAL

TO HAVE THE TECHNOLOGIES IN PLACE TO PERMIT THE LOW RISK INITIATION OF A RESEARCH STOVL SUPERSONIC FIGHTER/ATTACK AIRCRAFT IN THE EARLY TO MID-1990's

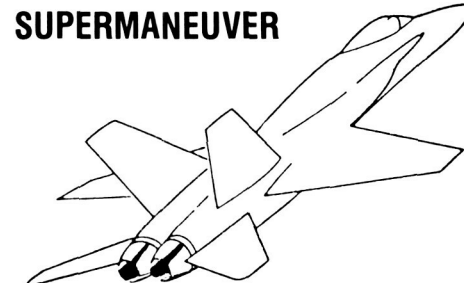
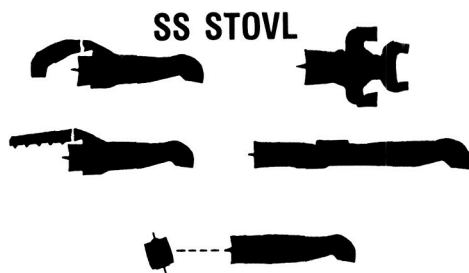
PROPULSION TECHNOLOGIES ARE THE KEY CRITICAL TECHNOLOGIES REQUIRED TO ACHIEVE THE GOAL. THIS IS A PROPULSION DRIVEN PROGRAM.

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PROPULSION TECHNOLOGIES

The propulsion technologies, identified as key to the supersonic STOVL program, cover a broad spectrum and are listed in the figure. Some of these are related to and will be developed in other on-going NASA and Air Force base technology programs combined with the STOVL program. The new higher T/W (15 to 20) engine core technology required will come from programs such as IHPTET and NASA's base technology efforts. Much high-alpha inlet and vectoring nozzle information will come from the NASA supermaneuver program with an F-18 aircraft at Dryden (HATP/HARV). The rest of the needed developments will be made in the supersonic STOVL program. The first issues to be resolved are the propulsive lift concepts themselves and which is the best propulsion system concept to pursue for a research aircraft. Each of the propulsive lift concepts has technical problems with performance, volume, weight, etc. We are actively addressing some of the key issues which will be shown shortly. A downselect will have to be made early in the program to manage the scope of this effort to appropriate levels. The impact of compressor bleed (for reaction control) will have to be evaluated for the higher T/W engines. New short diffuser inlets with high-alpha capability and vectoring nozzles with full 90° deflecting capability will have to be developed. Two of the more notable issues to be resolved include hot gas ingestion (HGI) and integrated flight/propulsion controls. The higher T/W engines will enhance the HGI problem already seen with the Harrier. Likewise, the propulsion controls will become more critical at takeoff, transition, and landing, where the traditional aerodynamic controls are relatively ineffective.

STOVL SUPERSONIC AND SUPERMANEUVER PROPULSION TECHNOLOGY



TECHNOLOGY ISSUES:

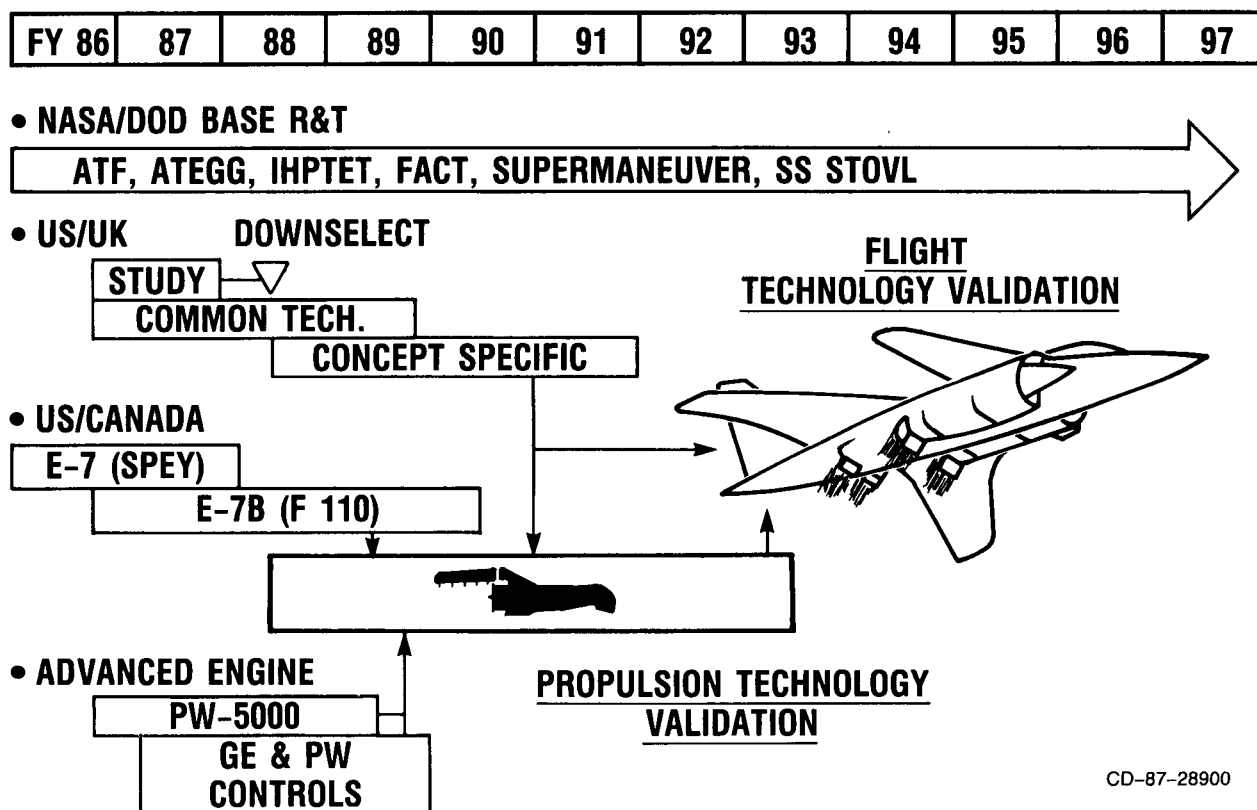
- PROPULSIVE LIFT CONCEPTS
- HIGH T/W ENGINES AND IMPACT OF ATTITUDE CONTROL SYSTEMS (BLEED)
- SUPERSONIC INLETS WITH HIGH ALPHA LOW SPEED CAPABILITY
- LIGHTWEIGHT MODULATING, DEFLECTING, AND VECTORING NOZZLES
- EFFICIENT LOW LOSS DUCTS, VALVES, AND COLLECTORS
- HOT GAS INGESTION AVOIDANCE/ACCOMMODATION
- INTEGRATED FLIGHT/PROPULSION CONTROLS

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SUPERSONIC STOVL PLAN

As stated in the program goal, there is interest in being able to develop and fly a research aircraft in the mid-1990's. The need for such an effort was clarified in the recent AF Forecast II study results which identified requirements for an aircraft with STOVL capabilities in the post-ATF time frame (beyond the year 2000). As previously stated, propulsion is key to achieving this. With that in mind, an enabling plan was developed and is shown in the figure. To meet the technology demonstration schedule, the required technologies have to be developed now, and a ground demonstration of the complete propulsion system for the research aircraft will have to take place early in the 1990's. As shown, several of the research programs have already been initiated. These include the NASA and DOD on-going base technology programs, the joint U.S./U.K. program, the U.S./Canada ejector program, and a series of contracted efforts with the major engine companies to investigate advanced engine concepts and, in particular, integrated flight/propulsion controls. These efforts include studies, experimental test programs, and some design (conceptual and detail) development efforts.

SUPERSONIC STOVL PROGRAM PLAN








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STOVL STUDY CONTRACTS





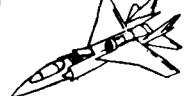
This figure presents a collage of the supersonic STOVL configuration study contracts currently in existence to generate the data necessary for the downselect process. This downselect process is now scheduled to begin early next year. Three propulsion system contracts, with General Electric (GE), Pratt and Whitney (PW), and Allison Gas Turbine (AGT) Division of General Motors, are being managed by Lewis. Also, four airframe contracts, with McAir, General Dynamics (GD), Grumman, and Lockheed, are being managed by Ames. Four of the engine concepts are being studied under the joint U.S./U.K. ASTOVL program and the fifth concept was added for consideration by NASA and the DOD to generate an appropriate data base with this configuration for comparison to the others. Three of the propulsion concepts were assigned to multiple contractors in order to generate comparisons. Each airframer was teamed with an engine company for each concept so that consideration of the joint requirements of each could be factored into the studies of each.

STOVL STUDY CONTRACTS—FY87

PROPULSION DATA BASE PREPARATION

| LEWIS/ENGINE CO. DATA BASE TASKS | US/UK | | | | NASA/DOD |
|--|--|---|--|--|---|
| |  VECT. THRUST |  EJECTOR |  RALS |  TANDEM FAN |  LIFT + LIFT/CRUISE |
| P&W ROLLS | X | | X | X(ROLLS) | |
| GE | X | X | X | | |
| AGT | | | X | X | X |



AIRFRAME/PROPULSION INTEGRATION

| AMES AIRFRAMER CONTRACTS | McAIR | GD | GRUMMAN | LOCKHEED | McAIR |
|---------------------------------------|---|---|---|---|---|
| |  |  |  |  |  |
| LEWIS/ENGINE CO. INTEGRATION TASKS | VECT. THRUST P&W X | EJECTOR GE X | RALS GE X | TANDEM FAN ROLLS (P&W) X | LIFT + LIFT/CRUISE AGT X |

BASE PROGRAM

The current Lewis base R&T program elements for the supersonic STOVL are shown in the figure. As indicated, these elements are focused on common technology issues. These are issues which would be applicable to two or more of the propulsion system concepts currently being studied in the supersonic STOVL technology program. The individual thrusts are either in existence today or are planned to begin shortly, as shown. Programs are already in progress for fan air collectors, valves, and ducting (for ejector and RALS systems); hot gas ingestion (HGI), short diffuser supersonic inlets with high-alpha capability; and integrated flight/propulsion controls. Each of these is important for all the systems. Information from each of the existing programs will be presented in the following figures. Plans are being developed to initiate in the near future corresponding programs in thrust augmentation by burning, and thrust deflecting and vectoring nozzles.

BASE PROGRAM

| COMMON TECHNOLOGY ISSUES | FISCAL YEAR | | | | | STATUS |
|---|---|----|--|----|----|----------------------------------|
| | 87 | 88 | 89 | 90 | 91 | |
| • FAN AIR COLLECTORS, VALVES, AND DUCTING (EJECTORS) |  | | | | | PROGRAMS IN PROGRESS |
| • HOT GAS INGESTION | | | | | | |
| • SHORT DIFFUSER SUPERSONIC INLETS WITH HIGH-ALPHA CAPABILITY | | | | | | |
| • INTEGRATED FLIGHT/ PROPULSION CONTROLS | | | | | | |
| • THRUST AUGMENTATION BY BURNING | | |  | | | PROGRAM PLANS BEING DEVELOPED |
| • THRUST DEFLECTING AND VECTORING NOZZLES | | | | | | |

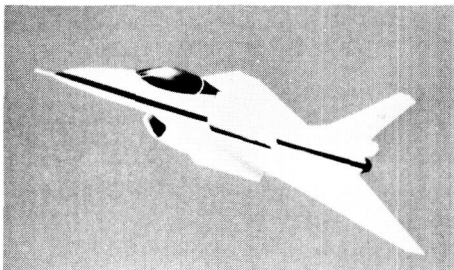
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U.S./CANADA EJECTOR TECHNOLOGY

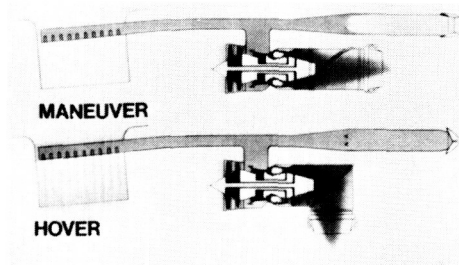
This figure is a collage of the various elements of the joint U.S./Canada ejector technology program. In this program, a full-scale model of the General Dynamics (GD) E-7 supersonic STOVL aircraft configuration will be tested in the Ames 40- by 80-foot wind tunnel. This aircraft incorporates the ejector augments propulsion concept to provide the required lift at takeoff and landing. The model will be tested with a complete engine system (first a Spey engine, then eventually with an F110). After the wind tunnel tests, the full-scale system components will be tested on the new Powered Lift Facility (PLF) at Lewis. Component performance tests can be run on this facility using high pressure (95 psig at 160 lb/sec) and high temperature (300 °F) combustion air for engine simulation. Both internal flow performance measurements and thrust performance can be determined on this facility. Eventually, testing with complete engine systems will be possible on the PLF. Results from the initial full-scale ejector tests on the PLF will be presented in the following figures.

US/CANADA EJECTOR TECHNOLOGY

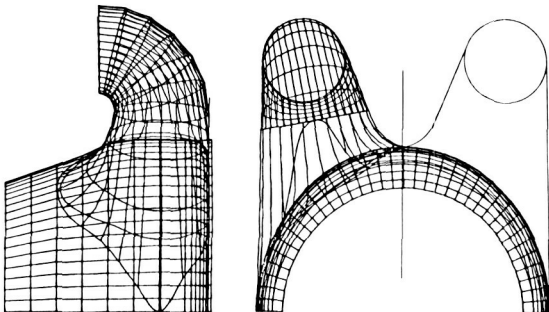
EJECTOR AIRCRAFT CONFIGURATION



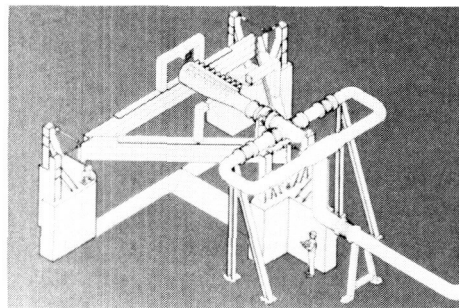
EJECTOR AUGMENTED LIFT SYSTEMS



FAN AIR COLLECTORS



POWERED LIFT FACILITY (PLF)

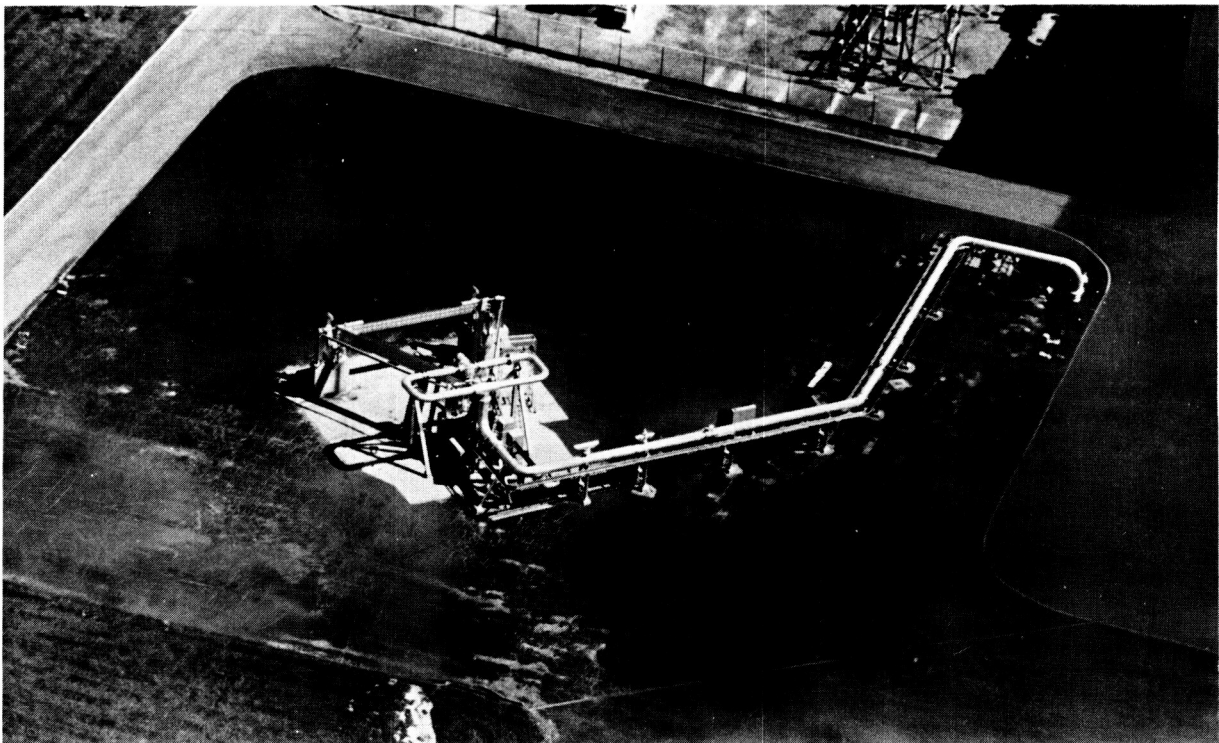


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POWERED LIFT FACILITY

The Lewis Research Center's new Powered Lift Facility (PLF), shown in the figure, was designed and built to support testing of all the concepts being proposed. The system includes a large (30 ft on a side) triangular frame supported 15 feet off the ground. This frame is supported by load cells, thereby providing a six-component force measuring system. Vertical (20,000 lb), axial (30,000 lb), and lateral (5,000 lb) forces can be measured in plus and minus directions. High pressure (95 psig) and heated (to 300 °F) air, at flow rates up to about 160 pounds per second, can be supplied to the stand to simulate fan exit conditions. The high pressure air is brought onto the system through a series of bellows, oriented 90° to the force system, to minimize momentum tare forces. The facility was completed and flow tests were initiated in September 1986. Initial force calibrations were made in April 1987, and performance tests began in June 1987. Plans are now being developed to modify the PLF for complete engine system testing.

POWERED LIFT FACILITY



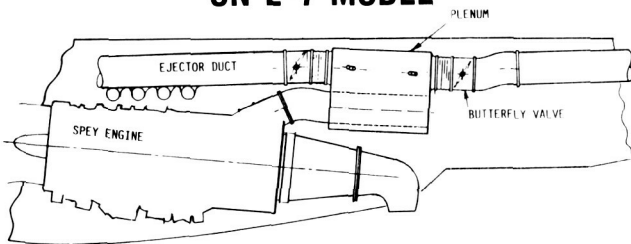
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deHAVILLAND FULL-SCALE DUCT AND VALVE TEST

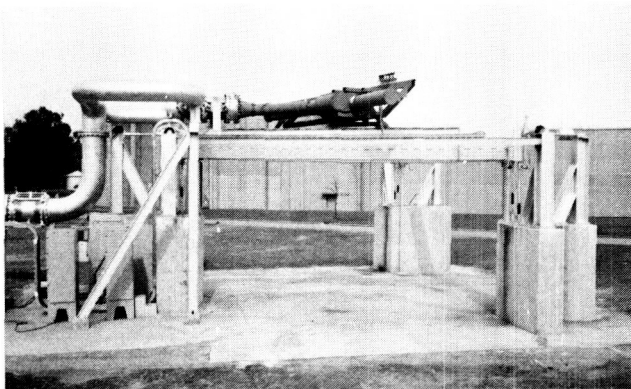
The first test on the new Powered Lift Facility (PLF) at Lewis was performed in September 1986. It included the full-scale internal fan flow ducting and valving scheduled to be installed in the full-scale GD E-7 aircraft model. A schematic of the installation on the aircraft model is shown in the figure. The fan flow will be collected and fed through a plenum to either a forward or aft duct. The forward duct will direct the flow to the ejector augmentor in the aircraft wing. The aft duct will lead to a thrust nozzle in the back. Flow direction will be controlled by butterfly valves. This arrangement is unique to the E-7 configuration. A photograph of the duct and valve hardware installed on the PLF is also shown. The purpose of these tests was to evaluate the pressure loss performance of the system before the ejector was installed. Typical pressure loss data are shown and compared to previous predictions (dashed curves). The test data show that the pressure losses were higher than predicted for the forward duct and lower than predicted for the rear duct. These results are expected to have minimal impact on the overall system performance.

deHAVILLAND FULL-SCALE DUCT AND VALVE TEST ON PLF

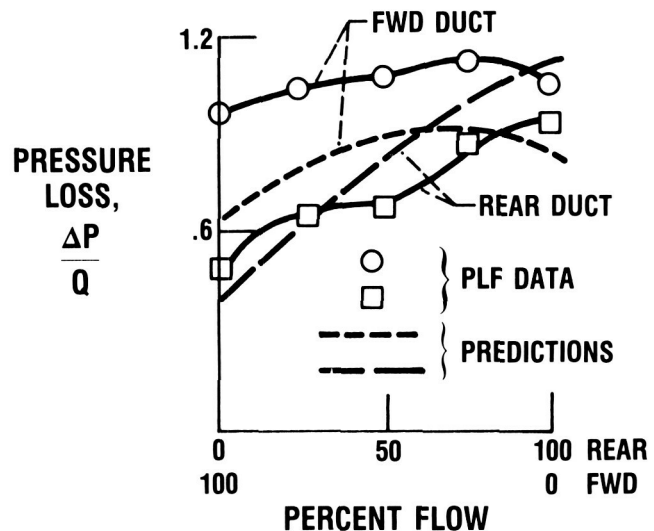
SCHEMATIC OF INSTALLATION ON E-7 MODEL



INSTALLED ON PLF



PRESSURE LOSS PERFORMANCE (SIM. FPR = 2.5)



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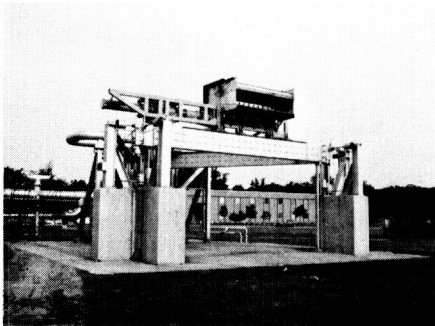
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deHAVILLAND FULL-SCALE EJECTOR TEST ON PLF

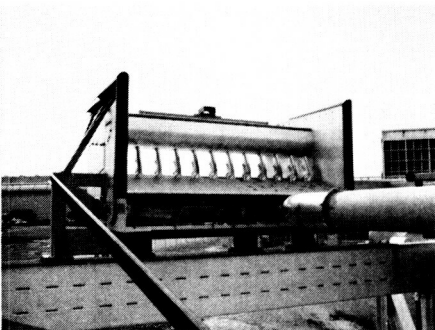
One half (side) of the full-scale ejector was attached to the system ducting and tested on the PLF in June 1987 for thrust performance. This figure includes an overall view of the model installed on the PLF and a closeup view. Preliminary thrust augmentation data are shown as a function of primary nozzle pressure ratio. The system design was for a thrust augmentation ratio of about 1.6 at a nozzle pressure ratio of 2.5. Previous deHavilland test results with another model and lower flow facility are also shown for comparison. The Lewis data show excellent agreement with the previous data and both exceeded the design by a considerable amount. The augmentation data shown are based on nozzle exit conditions. Correcting for the valve and duct pressure loss reduces this performance by only 3 percent. The resulting augmentation performance would still exceed the design requirement. This good agreement raises the confidence level for both the capabilities of the PLF and the feasibility of an ejector system as one of the viable concepts for a future supersonic STOVL.

deHAVILLAND FULL-SCALE EJECTOR TEST ON PLF

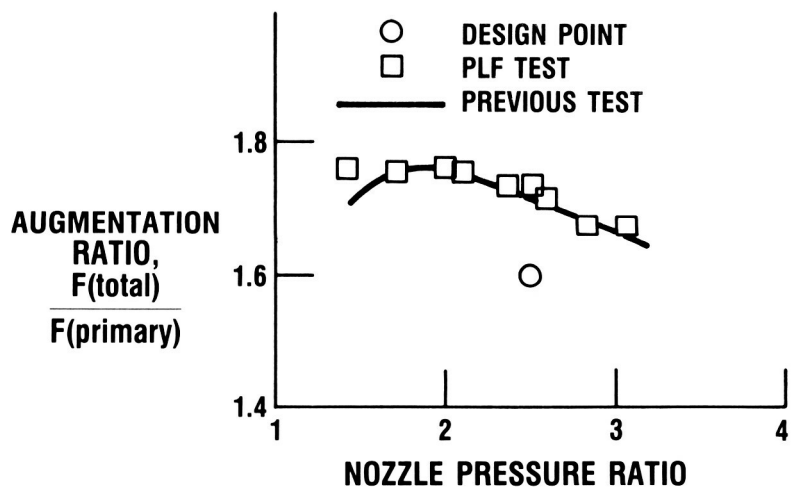
INSTALLED ON PLF



CLOSEUP VIEW



LIFT/THRUST PERFORMANCE



CD-87-28906

HOT GAS INGESTION (HGI)

Hot gas ingestion (HGI) will be a problem with all the proposed engine concepts. With the vectored thrust configuration, the problem results from the jet flow impinging upon the ground and (1) forming a fountain upwash which flows up to the fuselage and then forward to the inlet (near field) or (2) the ground flow feeding forward either interacts with the oncoming flow and gets recirculated or lifts off the ground because of bouyancy (far field). This has already been a problem for the Harrier. With the higher T/W engines required for the supersonic STOVL it will be a worse problem. Therefore either control devices or operational procedures will have to be developed to reduce or eliminate the problem. The following figures will describe the Lewis program in place to work the problem.

HOT GAS INGESTION (HGI)

STOVL DEFLECTED THRUST CONCEPT 279-3 AIRCRAFT

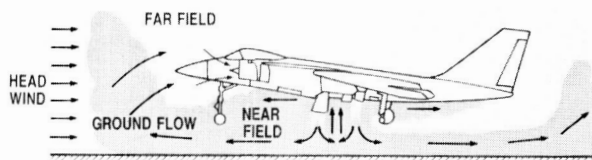


SOURCES

- NEAR FIELD
 - FOUNTAIN UPWASH
- FAR FIELD
 - SEPARATED GROUND FLOW

CD-87-28907

EXHAUST GAS INGESTION PHENOMENA



CONTROL

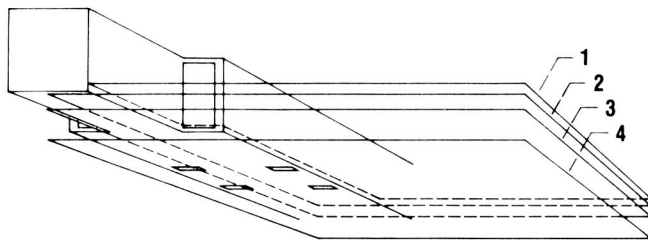
- LIFT IMPROVEMENT/FLOW DEFLECTOR
- INBOARD SPLAYING FRONT NOZZLES
- OPERATIONAL PROCEDURES
- ZONE BURNING FRONT NOZZLES
- INLET WATER INJECTION

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HGI ANALYTICAL RESULTS

An integral part of each of Lewis's programs includes analytical development. An example of where significant progress has been made is the program dealing with hot gas ingestion (HGI). Shown in the figure are preliminary results from calculations made using a 3-D Navier-Stokes Code based on the "TEACH" code of Imperial College. This code assumes incompressible gas, but allows temperature differences between gases and their corresponding different densities, so results at this stage are purely qualitative. Calculations were made around a simplified forebody/inlet configuration which included two subsonic jets close to a ground plane. A reflection plane down the middle of the fuselage provided an equivalent four-jet (vectored thrust), two-inlet configuration. The results shown are temperature profiles on planes at various heights from the ground to the aircraft inlets. As seen on the ground (plane 4), the jets show strong interactions, and the fountain upwash, typically seen with these configurations, is predicted along with the corresponding outflows and interactions with the free stream. Near the underside of the fuselage, a stronger forward flow is observed (plane 2), and then hot flow is actually seen entering the inlet (plane 1). Data from recent wind tunnel tests are being used to validate these results.

HOT GAS INGESTION



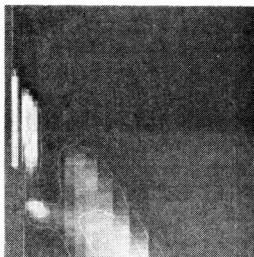
OBJECTIVE

**ASSEMBLE AND VALIDATE 3-D
COMPUTER CODES TO ANALYZE
EFFECTS OF AIRCRAFT CONFIGURATION,
FLIGHT SPEED, AND GROUND PROXIMITY
ON HOT GAS ENVIRONMENT AROUND
STOVL AIRCRAFT.**

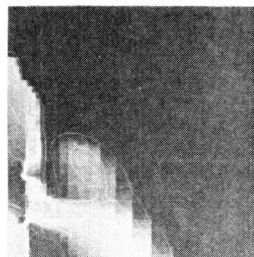
**FORWARD VELOCITY = 28 m/s
EXHAUST VELOCITY = 300 m/s**

TEMPERATURE CONTOURS

1000 F
70 F



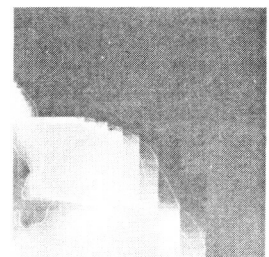
PLANE 1



PLANE 2



PLANE 3



PLANE 4

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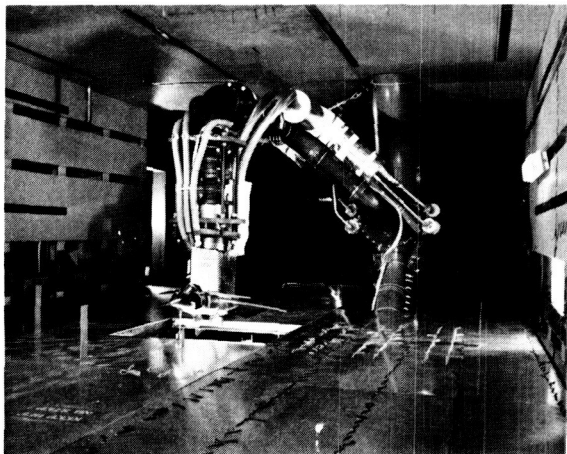
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ONE-TENTH SCALE HGI MODEL

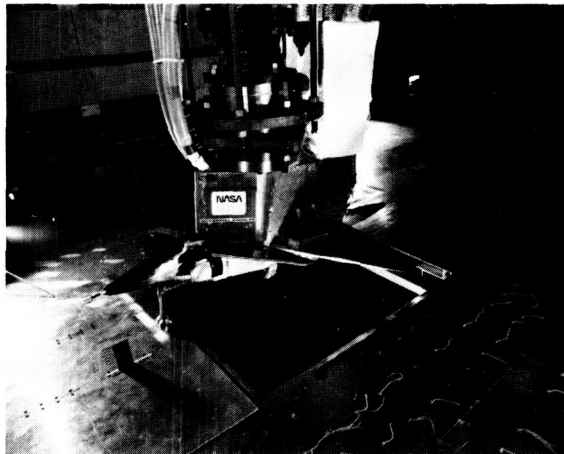
A 1/10 scale model of a McDonnell Aircraft Company (McAir) 279-3 supersonic STOVL aircraft configuration was tested in the 9- by 15-foot Low Speed Wind Tunnel (LSWT). This is a joint program between DARPA, NASA, and McAIR. The model is a four-nozzle, vectored-thrust configuration and includes high pressure, heated air (500 °F). Exhaust air provides inlet flow. Temperature and pressure rakes are included at the compressor face station to evaluate ingestion and determine distortion profiles. The model was mounted from a fairly rigid support which did provide some (but limited) model attitude and height variation. The tunnel installation included a ground plane which, as seen in the closeup view, included a trap door beneath the model. This allowed the hot gas to be ducted out of the tunnel while test conditions were being established. This door closed in about 0.5 second and then data were taken.

ONE-TENTH SCALE McAIR 279-3 SUPERSONIC STOVL MODEL HOT GAS INGESTION (HGI) TEST IN LeRC 9- BY 15-FT WIND TUNNEL

MODEL INSTALLED IN TUNNEL



CLOSEUP VIEW

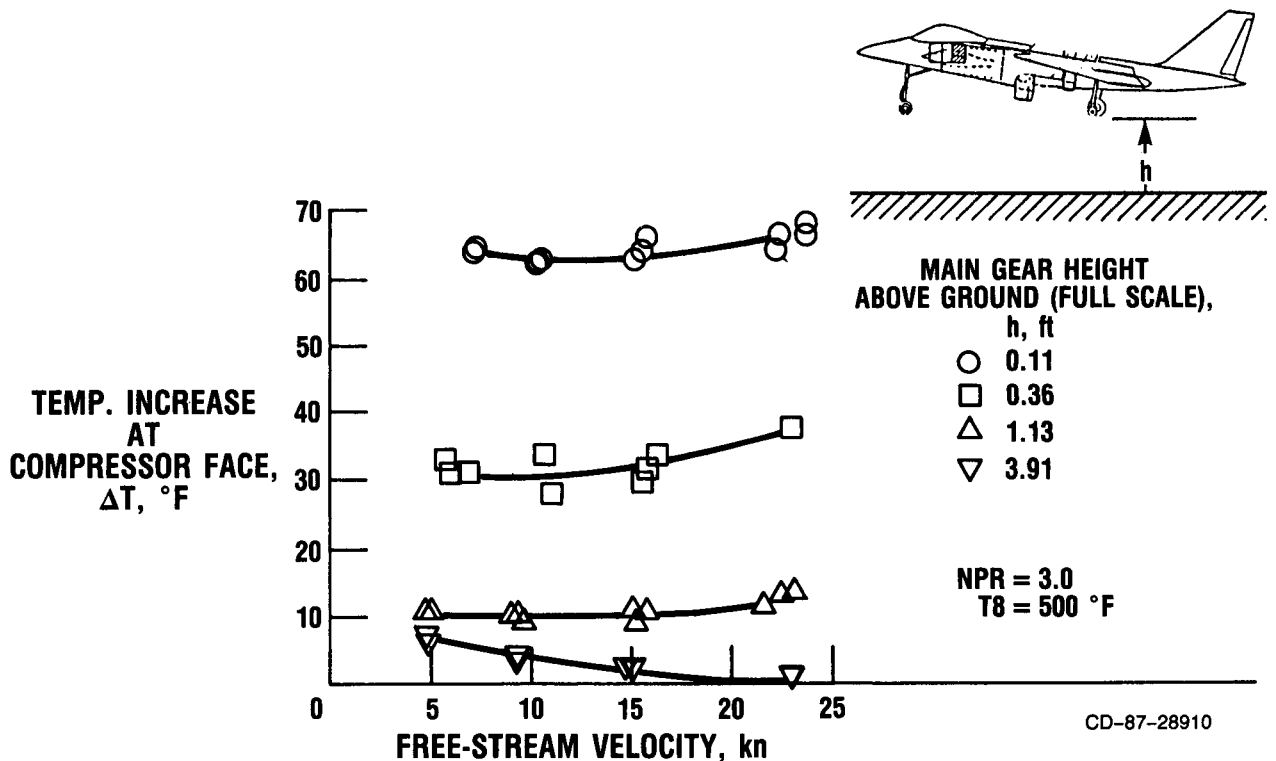


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EFFECT OF GROUND PROXIMITY ON HGI

Preliminary results of the effect of model height on inlet temperature are shown. As seen, hot gas began to be ingested with the landing gear, scaled up to full scale, about 4 feet above the ground for these simulated model and tunnel flow conditions. To use the model temperature on a full-scale basis it has to be scaled up, in this case, by a factor of about 4, since the nozzle supply was only at 500 °F. Investigation of temperature scaling was a part of this investigation. Preliminary results indicate that for some conditions the predicted scaling factors were validated but not for others. Basically, the trends observed in the data were as predicted. This test provided extremely valuable information to enhance the basic test technique for future tunnel entries and develop possible solutions.

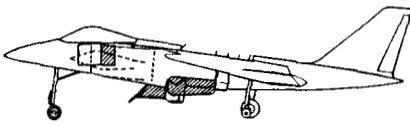
EFFECT OF GROUND PROXIMITY ON HOT GAS INGESTION (HGI) WITH BASIC 279-3 MODEL



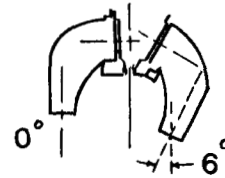
EFFECT OF MODEL GEOMETRY ON HGI

The data shown indicate that model geometry seems to have a more significant impact on HGI than either tunnel or model flow conditions. As seen, hot gas ingestion can be significantly reduced if the proper flow diverter were added to the model. In the testing, a series of supposed lift improvement devices (LIDS) were used. These devices were extremely effective in reducing the HGI. A change in nozzle splay angle could result in further reductions in HGI. It became apparent in this testing that each aircraft concept will probably be subject to HGI in some degree. However, there are many possible solutions, but the effectiveness of each will vary with the individual geometry.

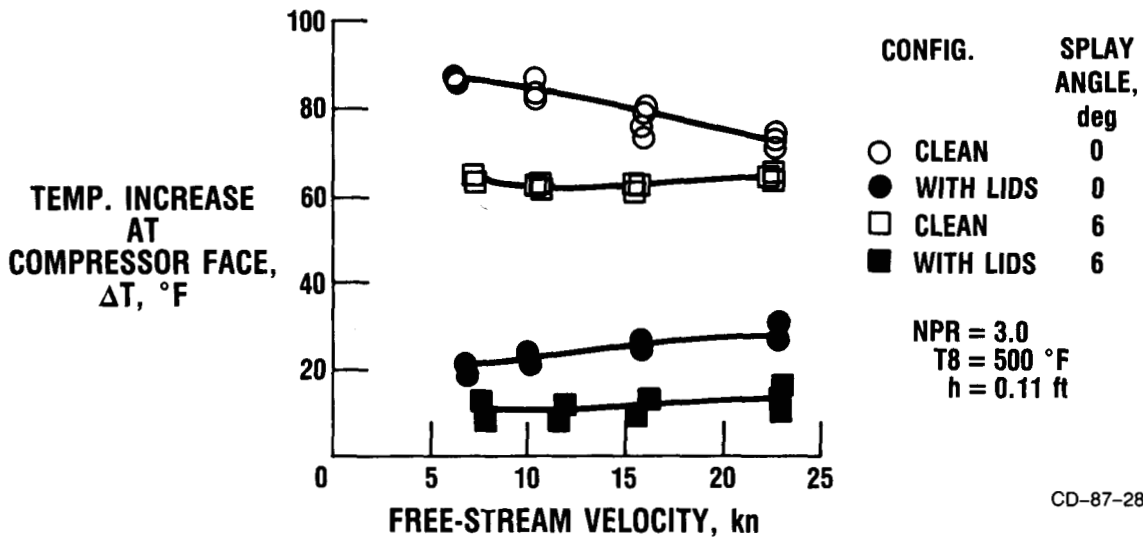
EFFECT OF GEOMETRY ON HOT GAS INGESTION (HGI)



LIFT IMPROVEMENT DEVICES (LIDS)



NOZZLE SPLAY ANGLE

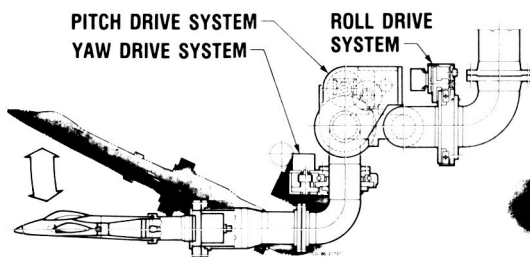


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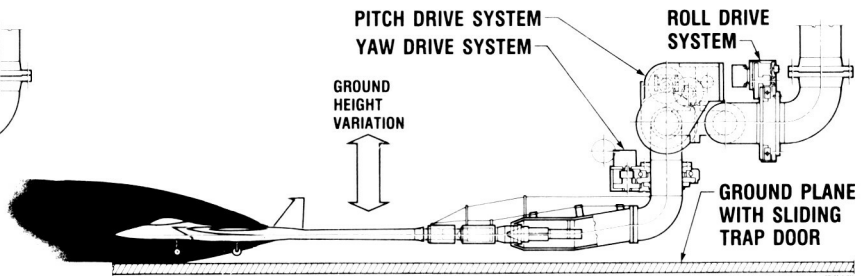
MODEL INTEGRATED SUPPORT SYSTEM (MISS)

As shown previously, the 1/10 scale model was mounted in the test section from a relatively rigid support system. For future tunnel entries, a new model integrated support system (MISS) is currently being fabricated. This support will allow testing at increased temperatures (1000°F) and have variable height, angle of attack, pitch, roll, and yaw. This support will again include exhaust for inlet flows and be capable of being used in other types of model testing. Thrust reverser, isolated inlet, and forebody inlet models could be tested over wide ranges of model and test conditions. The current model also will be modified to accept different nozzle configurations and locations. Both the model and MISS should be ready for a new series of tests in about a year.

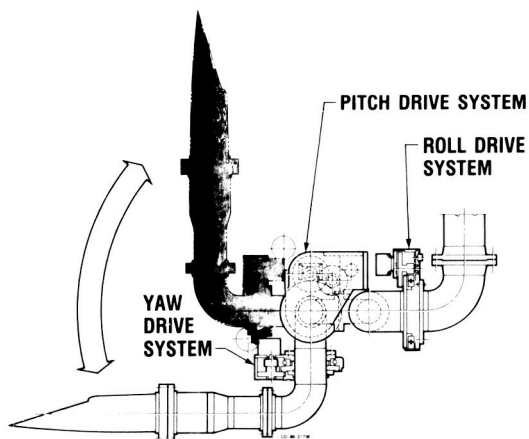
MODEL INTEGRATED SUPPORT SYSTEM (MISS) FOR VERSATILE AND EFFICIENT RESEARCH TESTING



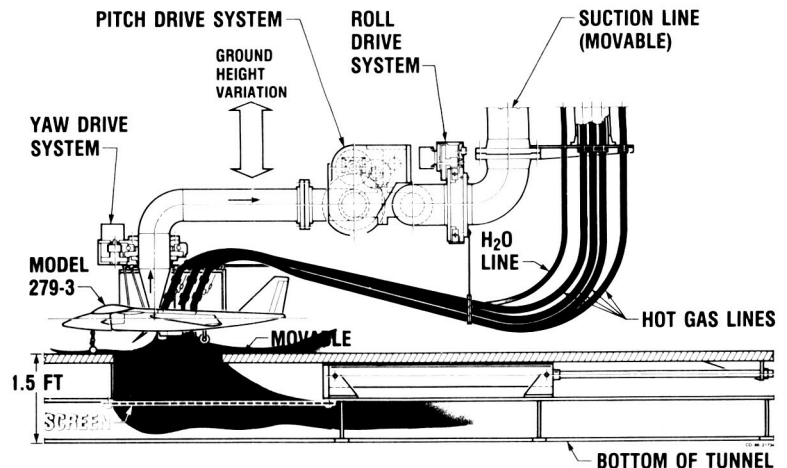
INLET-FOREBODY



STOL-THRUST DEFLECTION-REVERSING



ISOLATED INLET



ASTOVL-VERTICAL LANDING

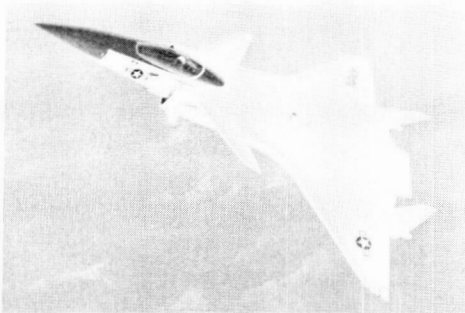
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SUPERSONIC INLET WITH SHORT DIFFUSER

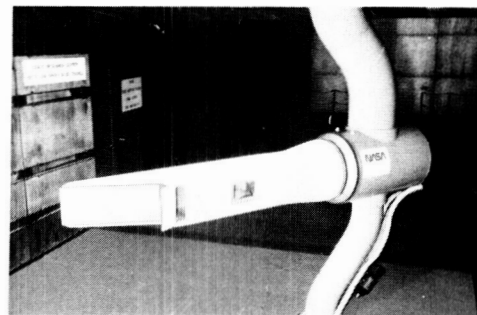
Conventional supersonic dash or cruise inlets have long diffusers which maintain well behaved, attached flows over wide ranges of aircraft angle of attack and attitude. The engine location in a typical supersonic STOVL may have to be moved forward for better weight and balance and to better locate the thrust vectors. This will then result in a problem for the diffuser design, particularly for operation at high angles of attack. A 2-D supersonic inlet model with a conventional-length diffuser was built and tested in the Lewis 9- by 15-foot Low Speed Wind Tunnel at angles of attack exceeding 100° . In these tests, variations in lip geometry and auxiliary inlets were investigated to improve angle-of-attack performance. Results of these tests show that good performance can be obtained even at the high angles of attack. A modification to this model has been designed and fabricated which includes a short diffuser more appropriate to STOVL configurations. Analysis has shown that this short diffuser will separate and have poor performance unless something is done to affect the boundary layer.

SUPERSONIC INLET WITH SHORT DIFFUSER

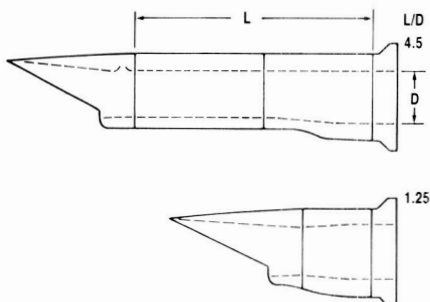
AIRCRAFT CONFIGURATION



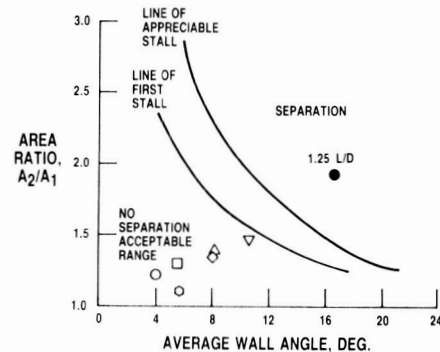
2-D SUPERSONIC INLET



DIFFUSER VARIATIONS



DIFFUSER FLOW RANGE



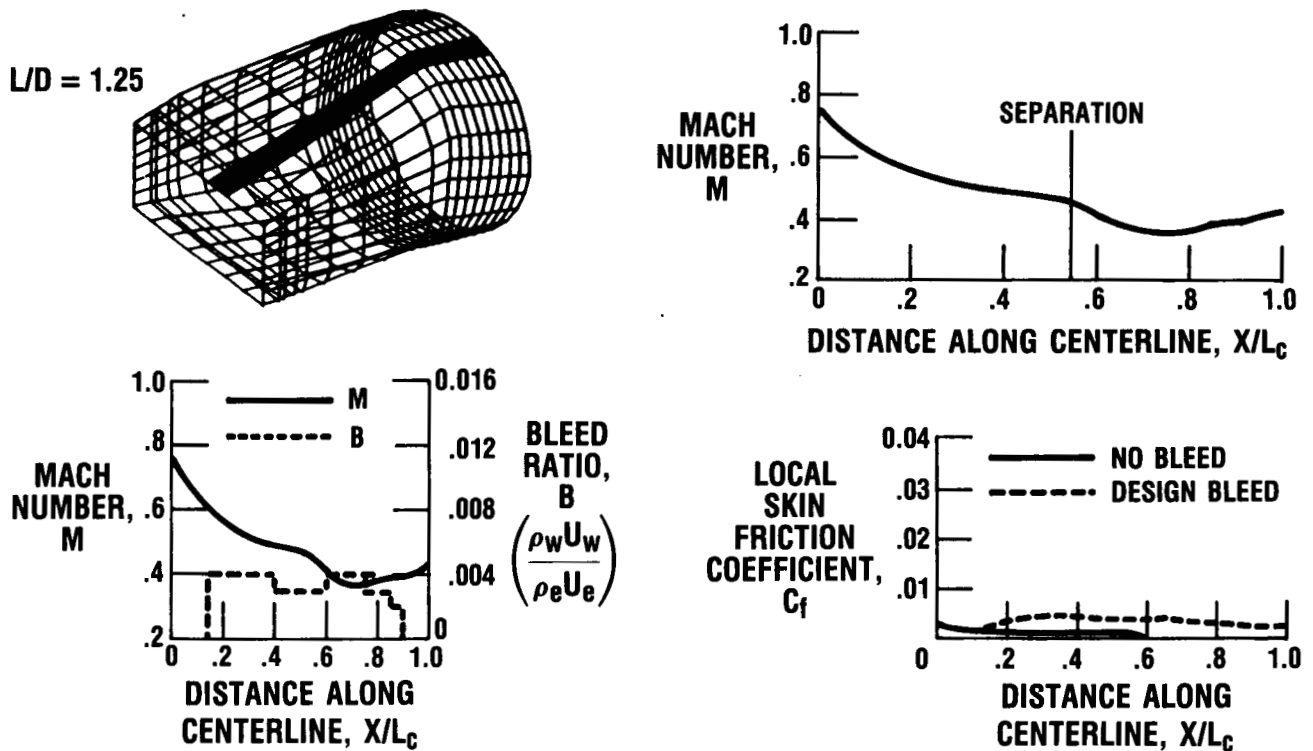
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SHORT DIFFUSER ANALYSIS

An example of the analysis used in the design of the new short diffuser is shown in the figure. This analysis used a methodology, developed at Lewis, which incorporates blowing for separated boundary layer control in subsonic V/STOL inlets at high angles of attack. The methodology was adapted by McAir to include natural bleed. The short diffuser ($L/D = 1.25$) was designed without bleed using typical techniques including potential flow codes and viscous corrections. Shown in the figure is the Mach number distribution along the top of the diffuser just outside the boundary layer. The analysis of this case indicated that the flow was separated about halfway back. This is reflected also in the skin friction, which approaches zero near this station. Varying distributions of boundary layer bleed were then analytically applied until this separation was eliminated (also reflected in the skin friction calculation). This result was achieved with reasonable amounts of bleed. Analyses were also made with jet blowing, again with favorable results. As a result of this work, a short diffuser model was designed which permits experimental incorporation of several different methods of boundary layer control, including suction, blowing (discrete and distributed), and other devices (e.g., vortex generators). The short diffuser model has been fabricated and is being readied for test in the Lewis 9- by 15-foot Low Speed Wind Tunnel. Data from these tests will be used to validate these analyses.

SHORT DIFFUSER ANALYSIS—EFFECT OF BLEED

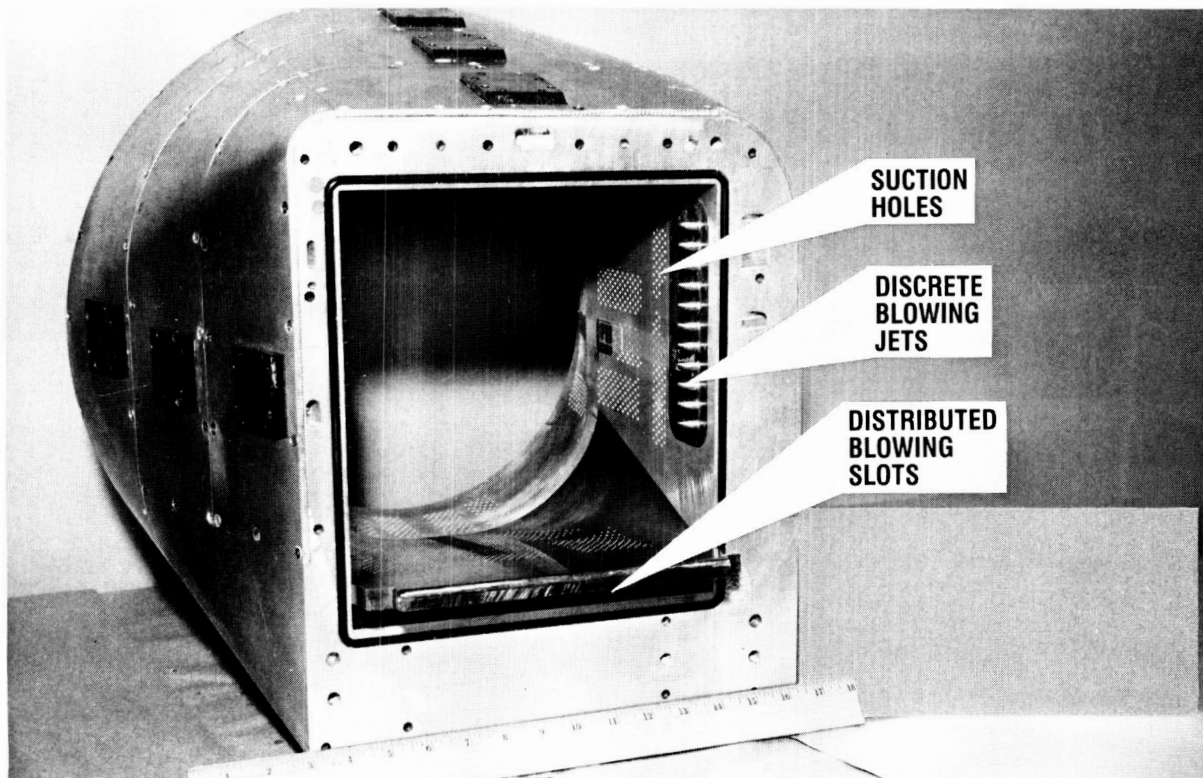


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SHORT DIFFUSER MODEL

This is a photograph of the short supersonic diffuser model. The model includes suction holes of varying porosity and distribution, discrete blowing jets for energizing the boundary layer, and distributed slots for blowing. The model can use any of these systems individually or in combination. Additional devices can also be used (e.g., vortex generators). This diffuser will fit into the previously tested inlet model and will be tested in the Lewis 9- by 15-foot Low Speed Wind Tunnel. Data will again be obtained up to angles of attack exceeding 100° .

SHORT SUPERSONIC DIFFUSER MODEL



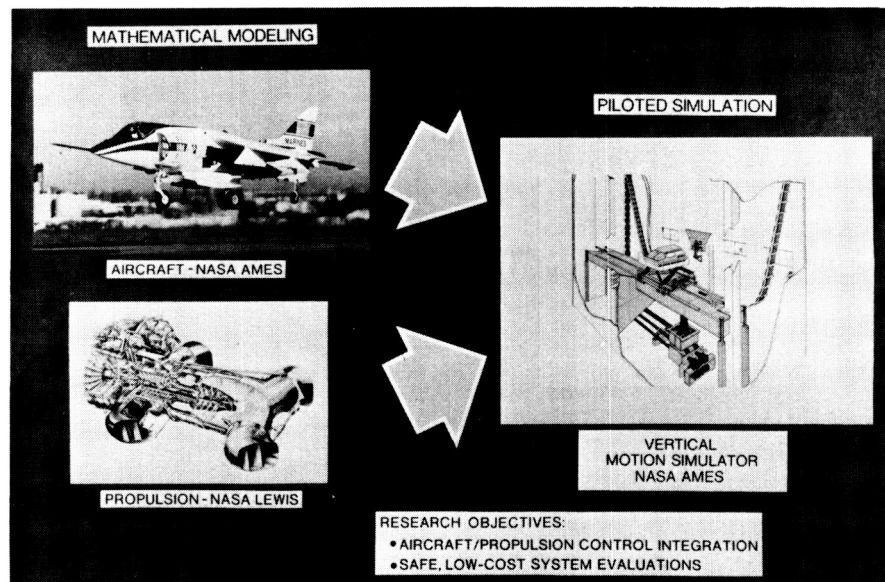
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INTEGRATED FLIGHT/PROPULSION CONTROLS

One of the most difficult technology issues relative to supersonic STOVL will be integrated flight/propulsion controls. In the joint U.S./U.K. program, this was identified as being critical enough to be started immediately. A joint effort was immediately organized between NASA Ames and Lewis to develop the required technology. This collage represents that joint effort. Ames will work the flight aspects, and Lewis the propulsion. Aircraft and engine simulations will be developed, and various control architectures will be pursued. A new hybrid computer has been installed at Lewis to pursue the propulsion simulations. The goals will be to eventually develop a pilot in the loop simulation capability, test these systems on the Ames Vertical Motion Simulator, and eventually verify the technology in a flight research program.

STOVL SUPERSONIC AND SUPERMANEUVER PROPULSION TECHNOLOGY

INTEGRATED FLIGHT/PROPULSION SIMULATION AND CONTROLS

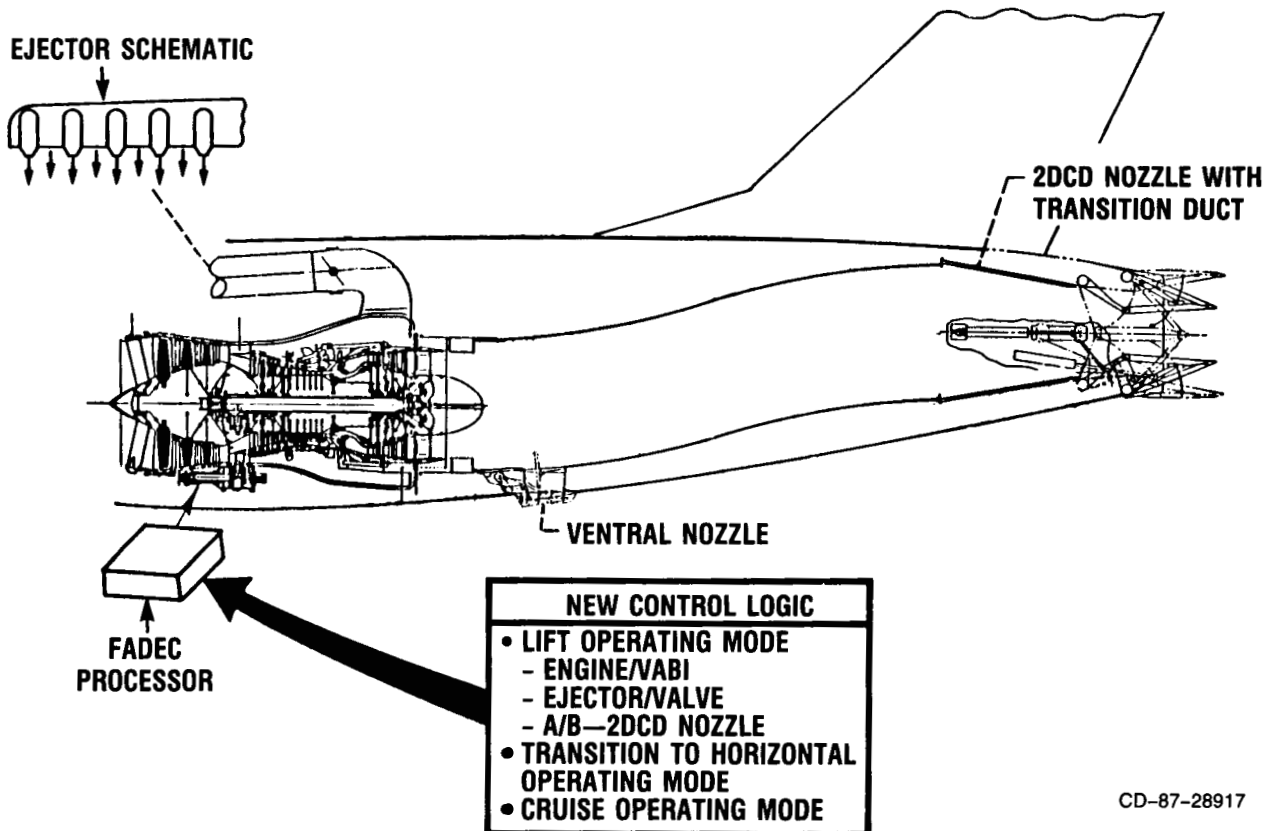


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EJECTOR/CONTROLS CONFIGURATION

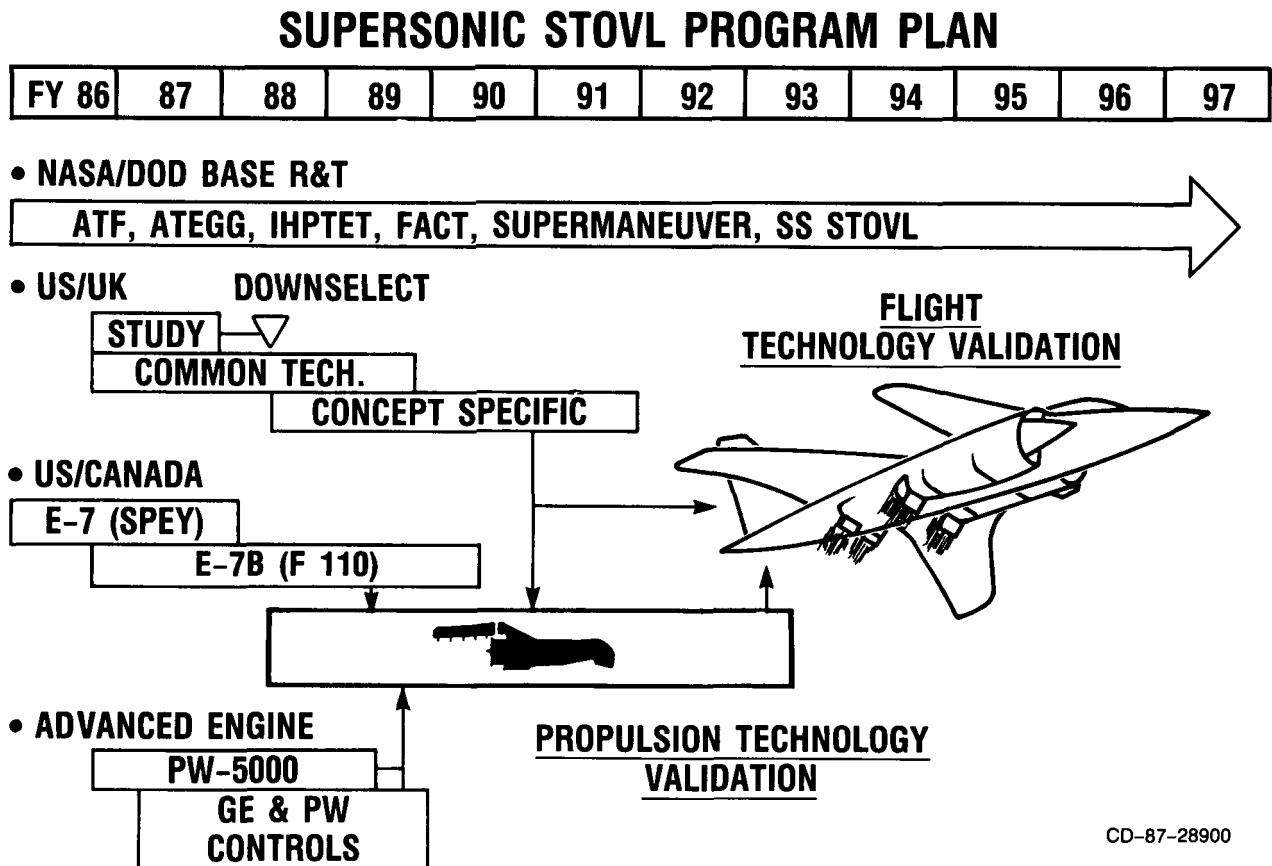
This figure is a schematic of the currently proposed ground engine and integrated flight/propulsion controls demonstration test. The configuration shown includes the GE F110 engine and ejector system which will be mounted on the Lewis Powered Lift Facility (PLF). The model will include a deflecting aft nozzle and a ventral nozzle for vertical thrust. The control computers will be fitted with the integrated control algorithms and a pilot station.

NASA/DARPA EJECTOR/CONTROLS CONFIGURATION



SUPERSONIC STOVL PLAN SUMMARY

In summary, a plan exists which will develop the required technology to allow the initiation of a research aircraft in the early- to mid-1990's. The interest seems to be there, as indicated by the involvement of three governments. The DOD is involved. The Air Force is already an active participant, and there is indication that the Navy will soon be involved. Successful studies and test programs are already in place and generating promising results.



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PROPULSION CHALLENGES AND OPPORTUNITIES FOR HIGH-SPEED
TRANSPORT AIRCRAFT

William C. Strack

ABSTRACT

E-3798-E

For several years there has been a growing interest in the subject of efficient sustained supersonic cruise technology applied to a high-speed transport aircraft. This presentation identifies the major challenges confronting the propulsion community for supersonic transport (SST) applications. Both past progress and future opportunities are discussed in relation to perceived technology shortfalls for an economically successful SST that satisfies environmental constraints.

A very large improvement in propulsion system efficiency is needed both at supersonic cruise and subsonic cruise conditions. Toward that end, several advanced engine concepts are being considered that, together with advanced discipline and component technologies, promise at least 40-percent better efficiency than the Concorde engine.

The quest for higher productivity through higher speed is also thwarted by the lack of a conventional, low-priced fuel that is thermally stable at the higher temperatures associated with faster flight. Extending Jet A-type fuel to higher temperatures and the adoption of liquefied natural gas (LNG) or methane are two possibilities requiring further investigation.

Airport noise remains a tough challenge because previously researched concepts fall short of achieving FAR 36 Stage III noise levels. Innovative solutions may be necessary to reach acceptably low noise.

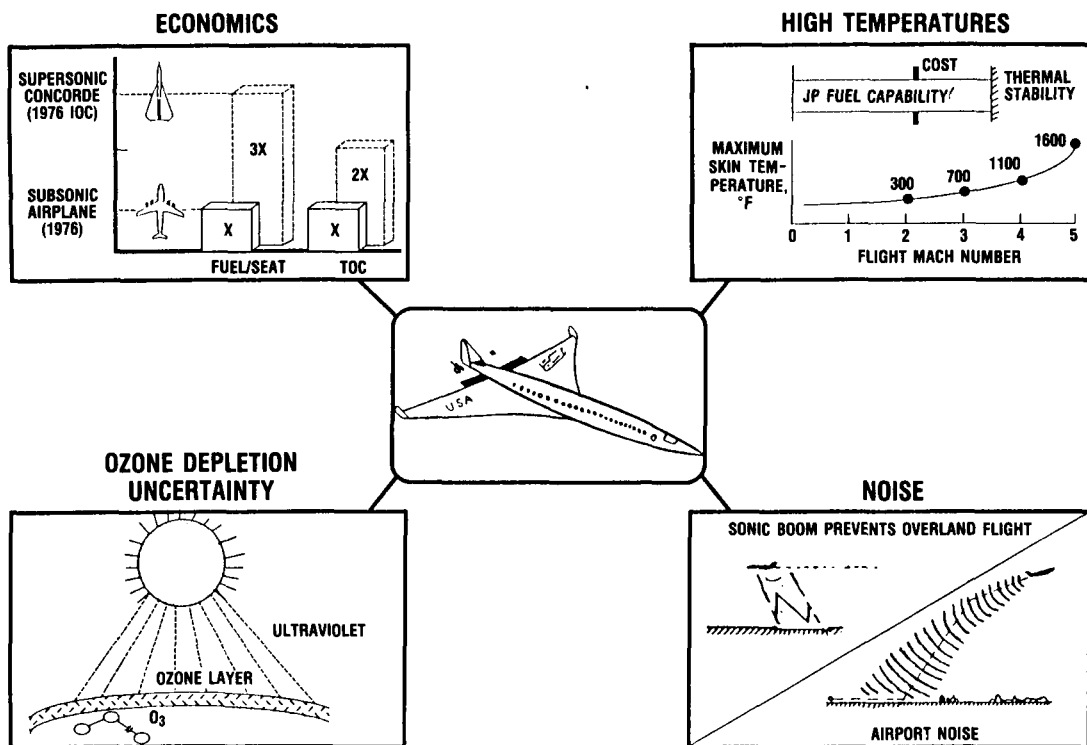
While the technical challenges are indeed formidable, it is reasonable to assume that the current shortfalls in fuel economy and noise can be overcome through an aggressive propulsion research program.

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CHALLENGES TO HIGH-SPEED TRANSPORTS

Although the Concorde ushered in the supersonic transport era, it has not been a commercial success for a variety of reasons. Its poor fuel consumption (3 times equivalent technology subsonic airplanes) is largely responsible for its uncompetitive economics; the total operating cost (TOC) is twice that of similar technology, long-range subsonic transports. Very large airframe and propulsive efficiency improvements will be required to alter this situation. In our quest for greater productivity through increased speed, we are confronted with an ever increasing technical challenge arising from high ram temperature levels. In addition to airframe skin temperature problems, the inability of readily available, low-cost fuels to provide adequate thermal stability seriously impedes the pursuit of higher speeds. Expensive JP-type fuels reach thermal stability limits at approximately Mach 3-1/2, but low-cost Jet A is limited to only Mach 2+. While both sonic boom and airport noise levels are currently excessive, only the airport noise problem is of primary concern to the propulsion industry. Another potential environmental issue is the depletion of atmospheric ozone via jet engine exhaust-gas emissions.

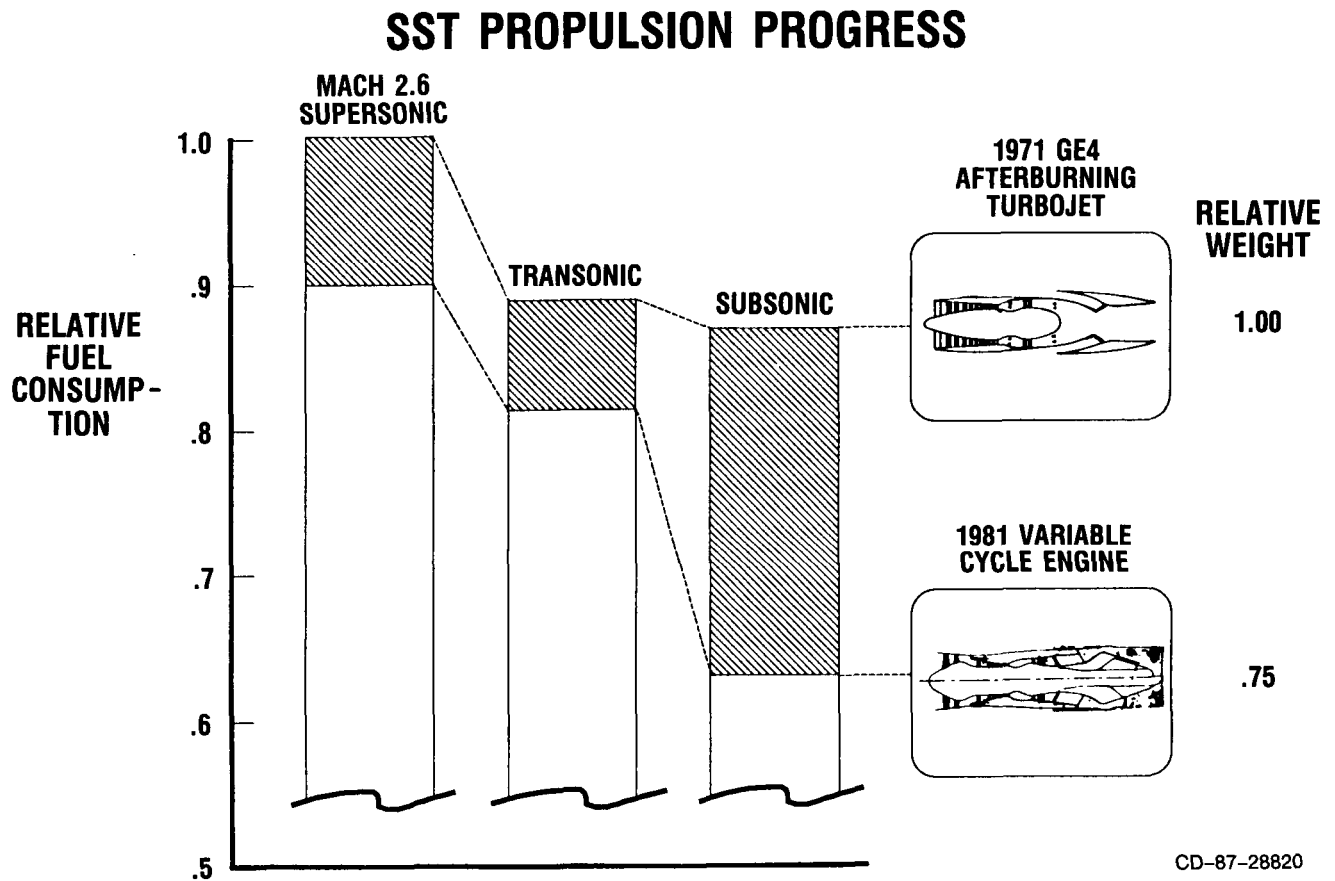
CHALLENGES TO HIGH-SPEED TRANSPORTS



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SST PROPULSION PROGRESS

Considerable progress was achieved during the 1970's in the NASA-sponsored variable-cycle engine (VCE) program. Compared to the 1971 GE4 afterburning turbojet (ABTJ), the 1981 VCE's consumed 10 percent less fuel at supersonic and transonic conditions, and 25 percent less at subsonic speeds -- reflecting the cycle-changing feature of VCE's. A simultaneous 25 percent reduction in engine weight occurred. Nevertheless, these gains are insufficient by themselves to enable competition with subsonic aircraft. The subsonic efficiency of the 1981 VCE engines, for example, is still only one half that of today's high bypass-ratio turbofans.

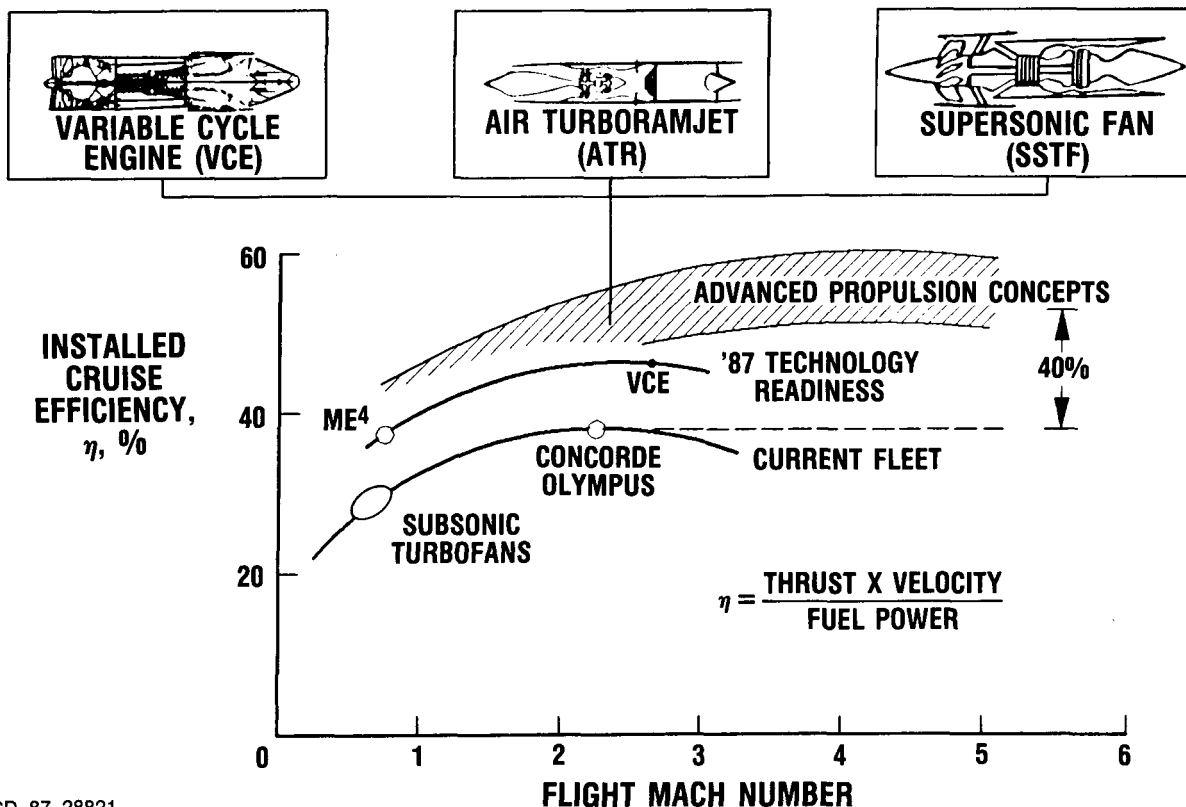


FUTURE HIGH-SPEED PROPULSION PERFORMANCE POTENTIAL

The primary cause of the Concorde's high fuel consumption is the dramatic fall in airplane lift-to-drag ratio (L/D) at supersonic speeds which is on the order of 1/2 that of subsonic transports. This is only partially offset by the trend toward increasing overall engine efficiency with flight speed. "Installed cruise efficiency" shown here includes inlet and nozzle losses, but not nacelle drag, and represents design point values. The middle curve indicates that significant improvement is possible with today's available technology for both subsonic (maximum efficiency - E³ technology) and supersonic regimes.

The top band projects future opportunities based principally on NASA cycle analyses. Several alternative cycle concepts are represented, including very advanced VCE and turbine bypass engines (lower boundary), and radically different concepts such as regenerative air turboramjets (ATR's) and supersonic throughflow (SSTF) turbofans (upper boundary). These advanced technology concepts extend the peak propulsion-efficiency levels from Mach 2+ to at least Mach 4. Gains of 40 percent or more over Concorde's Olympus are possible. Using a simple criterion such as design point efficiency is insufficient to properly convey overall impact. For example, this plot shows a relatively modest 8-percent gain between 1987 technology VCE's and advanced VCE's (lower line of top band). Not shown, but also important are even larger gains in climb efficiency and weight for advanced VCE's.

FUTURE HIGH-SPEED PROPULSION PERFORMANCE POTENTIAL



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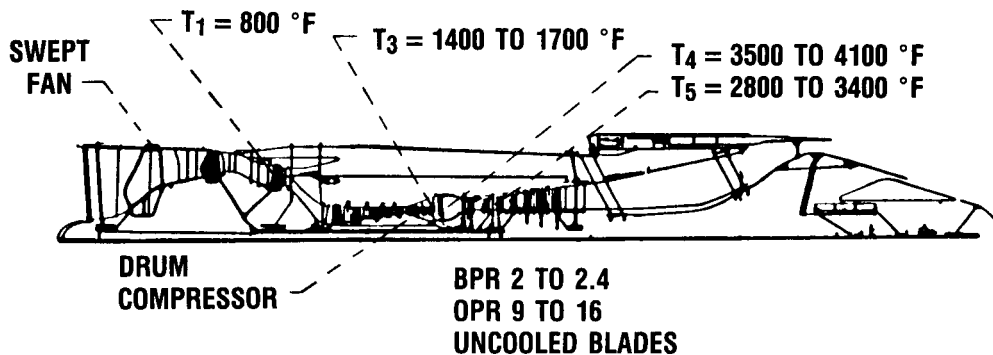
VARIABLE-CYCLE ENGINE GOAL

The most obvious contender for a future SST is an advanced variable-cycle engine. This approach builds on the previous VCE philosophy of mitigating the off-design compromises inherent in a fixed-geometry engine. This is accomplished by incorporating enough variable geometry features to yield respectable performance over a wide range of flight speeds and power settings.

Displayed here is an example of a "goal" VCE, representing what payoffs would accrue if revolutionary advances in materials and structures technology are achieved. This particular design was generated by General Electric in their recent NASA-sponsored Revolutionary Opportunities for Materials and Structures (ROMS) study. It assumes essentially uncooled stoichiometric engine materials coupled to advanced aerodynamics and structural design technologies. This implies extensive use of nonmetallics and intermetallic materials.

Two levels of technology are quoted here: (1) the full stoichiometric goal level is denoted by the right-hand values (GE ROMS), and (2) a 600 °F cooler level is denoted by the left-hand values (NASA estimate). One-third of the 28-percent fuel reduction is due to a 45-percent engine weight reduction relative to a hypothetical 1984 technology-readiness baseline engine.

VARIABLE-CYCLE ENGINE GOAL POTENTIAL MACH 3 CRUISE CONDITIONS



**BENEFITS (MATERIALS AND AERO) : 290 PAX 5000 nmi TRANSPORT
RELATIVE TO CURRENT TECHNOLOGY AT \$1.00/gal.**

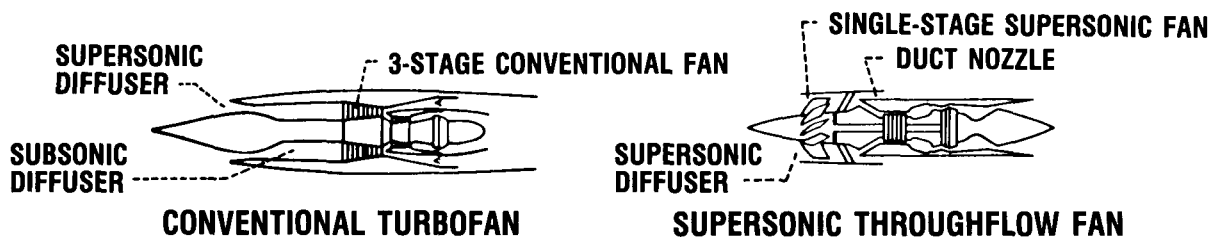
| | |
|-------------|------------------|
| FUEL | 24 TO 28% |
| DOC | 17 TO 20% |

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SUPERSONIC THROUGHFLOW FAN ENGINE

One potential SST breakthrough is the supersonic fan concept. Instead of using a long and heavy inlet system to efficiently decelerate the intake airflow to the subsonic speeds required by conventional turbomachinery, the supersonic fan efficiently processes air at supersonic throughflow velocities. The advantages include much lower inlet-system weight, lighter fan (less stages required for a given pressure ratio), less boundary-layer bleed drag, better inlet pressure recovery, and better matching of bypass ratio variations to flight speed (M_0). Of course, there are many unknowns and challenges. What are such a fan's low-speed operating characteristics? How can the core inlet losses associated with unsteady, swirling, supersonic inflow be controlled; or is an aft fan configuration a better solution? Little effort has been expended on this concept to date, although NASA has initiated a concept feasibility research effort.

SUPERSONIC THROUGHFLOW FAN ENGINE



SUPERSONIC THROUGHFLOW FAN ENGINE FEATURES

- SHORT, ALL SUPERSONIC INLET
- SINGLE-STAGE SUPERSONIC FAN
- BPR DECREASES WITH M_0

IMPLICATIONS

- LOWER WEIGHT, LOWER INLET DRAG
- LOWER WEIGHT AND COST, RUGGED BLADING
- HIGHER CRUISE THRUST

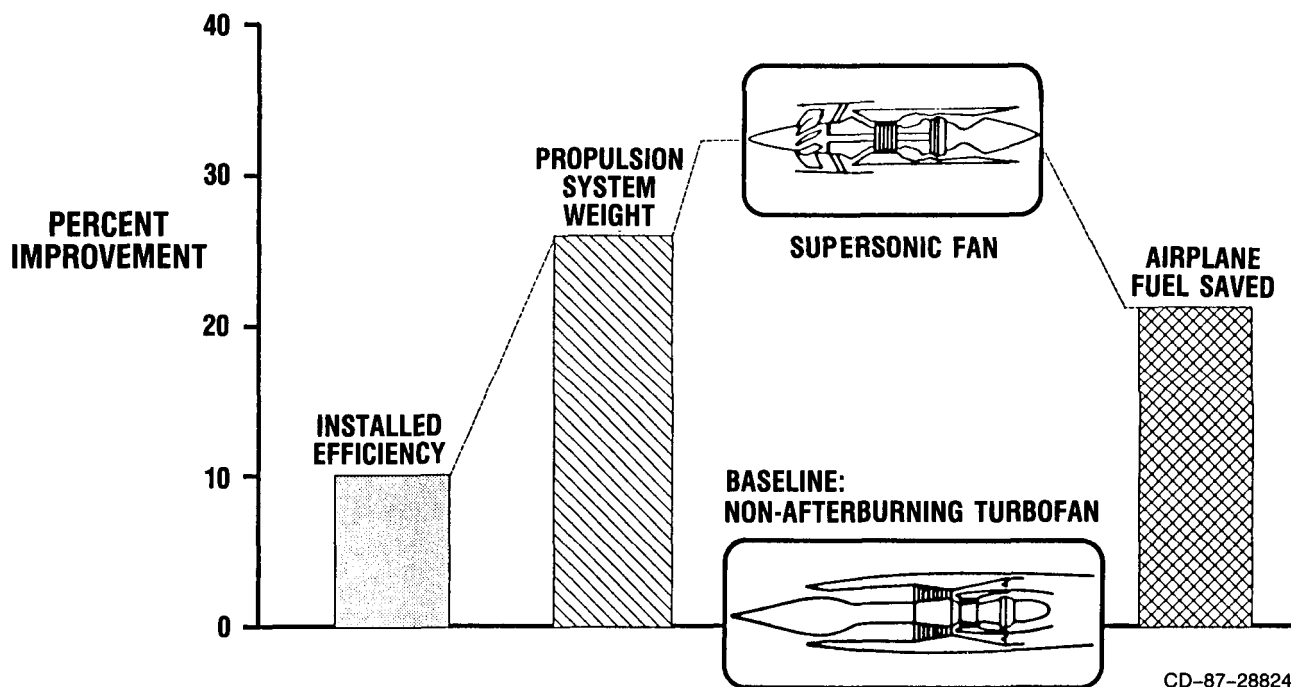
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BENEFIT OF SUPERSONIC THROUGHFLOW FAN

The potential payoff of supersonic throughflow fan (SSTF) technology for a typical SST application has been analyzed by NASA in-house (NASA TM-100114). One of the major contributors is the inlet size and weight reduction to about 1/2 that of a conventional supersonic inlet. This also reduces the inlet bleed-drag penalty. Furthermore, the higher SSTF inlet recovery leads to more thrust/airflow at cruise, and less transonic-spillage drag when external compression inlets are used. The 35-percent larger cruise thrust/airflow could mean a smaller engine is required dependent on the engine-sizing criteria. In the payoffs quoted here, takeoff thrust/weight was held fixed to maintain good takeoff performance.

BENEFIT OF SUPERSONIC THROUGHFLOW FAN

**MACH 3 COMMERCIAL TRANSPORT
300 PASSENGERS, 5500 nmi RANGE**

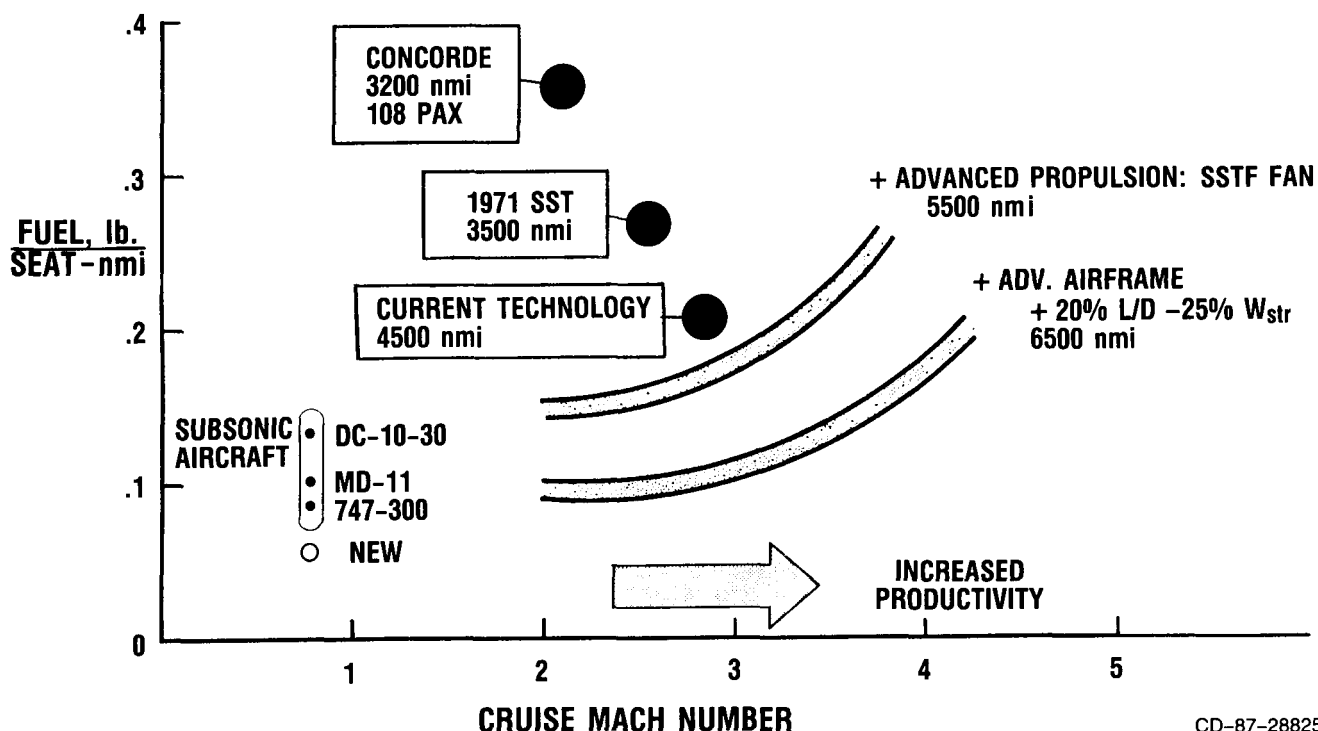


ADVANCED SUPERSONIC TRANSPORTS COULD ACHIEVE COMPETITIVE FUEL ECONOMY

This chart displays the impact of potential future technology advances on airplane fuel consumption while recognizing that the key to viable SST economics is fuel cost levels approaching those for future subsonic airplanes. Achieving 100-percent fuel-usage parity with the subsonic competition is not necessary because of the increased productivity associated with SST's. However, it is important to at least be in the same neighborhood, which the Concorde and previous SST-study airplanes cannot achieve despite their shorter ranges. The impact of advanced propulsion technology is impressive, enabling fuel-consumption rates not much different than current long-range subsonic airplanes. Coupling the most optimistic propulsion technology with potential airframe advances in L/D and structural weight (W_{str}) produces encouraging results in the Mach 2 to 4 range. Of course, these are preliminary, first-order results subject to refinement as the ongoing studies evolve. Another uncertainty is the possible introduction of a very advanced, all-new subsonic airplane. An estimate of that possibility is included here that has an 11-percent L/D improvement, a 15-percent structural weight improvement, and a 33-percent propulsion-efficiency improvement. The conclusion to be drawn from this analysis is that the SST fuel-consumption impediment can be overcome, but it will require very large technology gains in all disciplines -- propulsion, aerodynamics, and structures.

IMPACT OF TECHNOLOGY ON FUEL ECONOMY

300 PASSENGERS

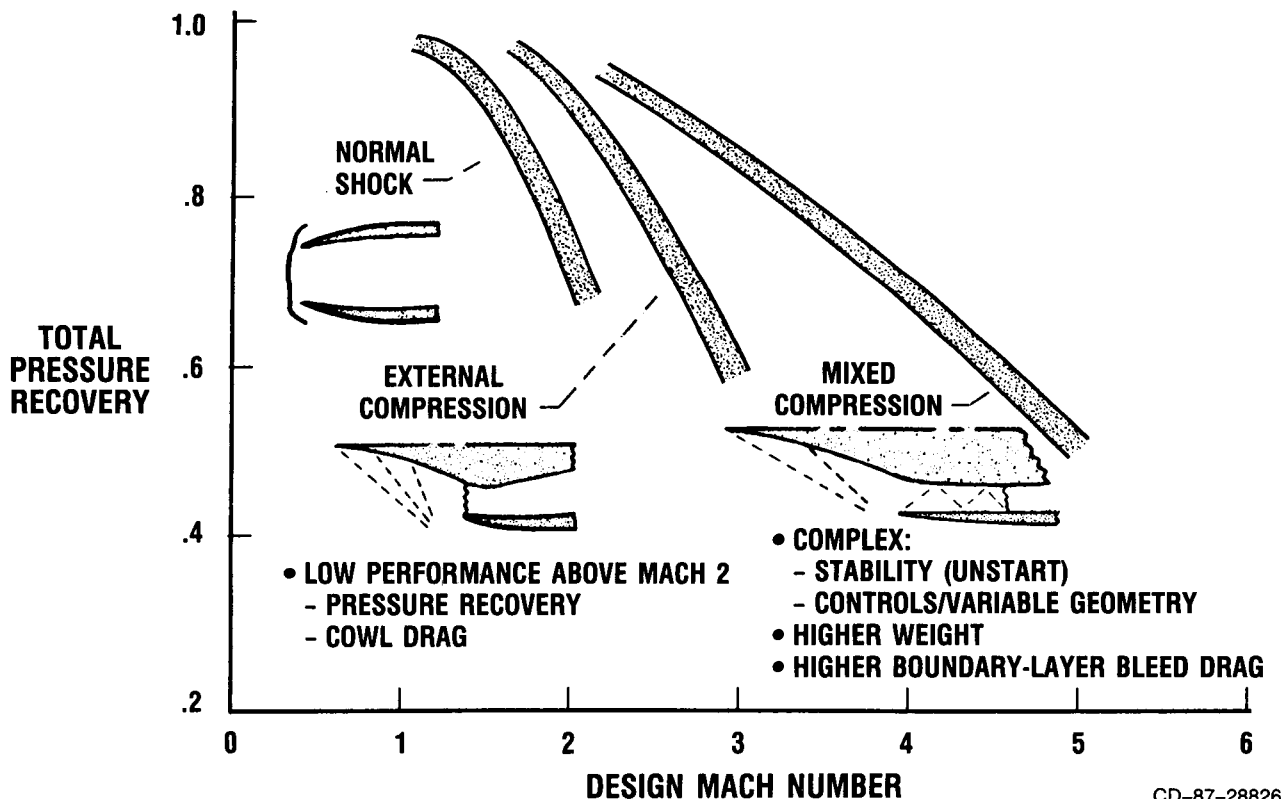


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SUPERSONIC INLET PERFORMANCE

Commercial supersonic flight at Concorde speeds (Mach 2) can be viewed as relatively straightforward and within industry's technological grasp. Pushing the cruise speed substantially higher is certainly desirable, but introduces a series of ever-increasing technological challenges. One of these new challenges is illustrated here. Conventional external compression inlets accomplish all of their diffusion outside of the intake duct through several oblique shocks and a terminal normal shock located at the cowl lip. This type of inlet delivers adequate performance and is well-behaved (stable) under all transport flight conditions up to Mach 2. Beyond Mach 2 though, the performance of external compression inlets rapidly deteriorates because of the excessive cowl drag associated with the increasing cowl-lip angle and the inability to increase the number of oblique shocks because of excessive inlet length and weight penalties. Flight beyond Mach 2, therefore, requires a mixed-compression-type inlet that performs some of the diffusion inside the intake duct through more oblique shocks and a normal shock near the throat. This introduces other problems: notably, more boundary-layer bleed to avoid adverse shock-boundary-layer interactions (separation) and inlet shock-system instability. The result is a much more complex inlet and control system. Neither transports nor fighters have been flown operationally with such inlets, yet the need for utmost propulsion efficiency will require it for high-speed transports.

SUPERSONIC INLET PERFORMANCE

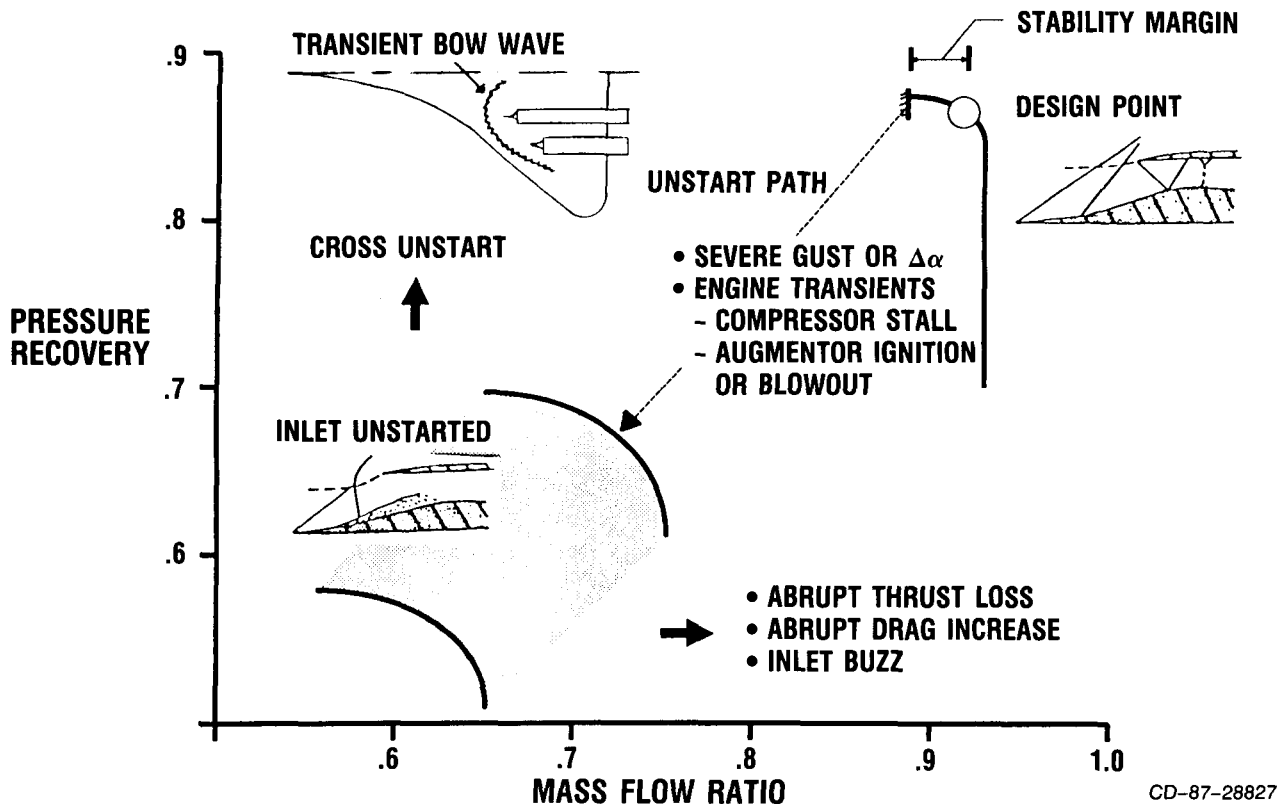


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MIXED-COMPRESSION SUPERSONIC INLET INSTABILITY

Mixed compression inlets are quite susceptible to a phenomenon known as inlet instability or "unstart." Whenever a flow-retarding disturbance occurs, the internal shock system moves abruptly upstream and repositions itself completely outside the intake duct. This causes an abrupt and severe drop in thrust due to lower recovery and mass flow, and an increase in drag. The precipitating disturbance could be relatively small, such as encountering a strong gust or rapidly changing the angle-of-attack. If the initial disturbance is large (e.g., compressor stall), the transient response can be very severe -- possibly unstating neighboring inlet-engine systems which would likely throw the airplane into a violent yaw and roll maneuver. To prevent such undesirable behavior, some form of stability control system is needed.

MIXED COMPRESSION SUPERSONIC INLET INSTABILITY

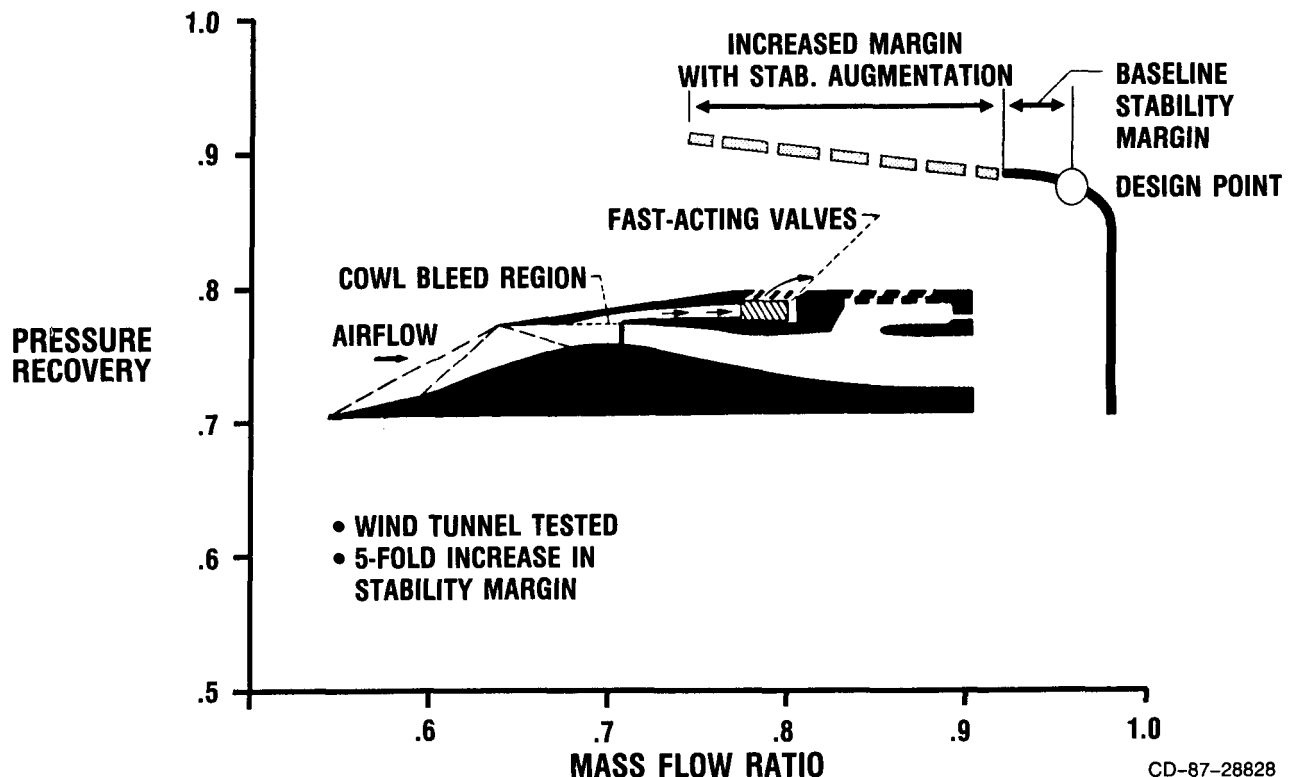


MIXED-COMPRESSION INLET STABILITY IMPROVEMENTS

This inlet stability improvement concept consists of a set of self-actuating bleed valves located in the inlet nacelle. These rapid-response-rate pneumatic valves will open in response to the increase in duct pressure produced by a transient excursion of the inlet terminal shock forward from its steady-state position. As the shock moves forward it exposes the stability bypass plenum to increased pressure and automatically activates the bleed valves which spill inlet bleed air overboard. This increases the inlet mass flow and forces the shock rearward, and thereby reestablishes stability. The valves close when the transient disturbance subsides and the shock has retreated to its steady-state position.

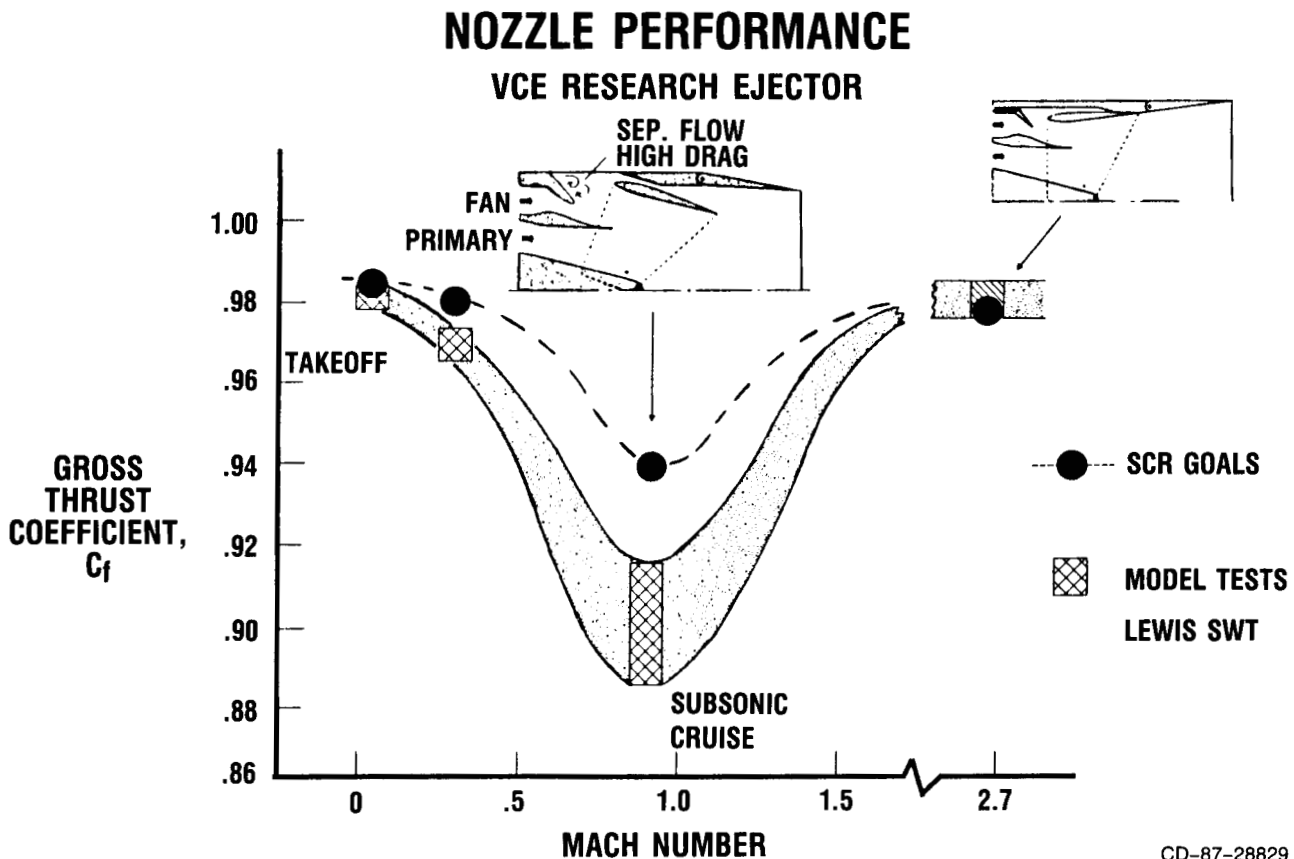
An experimental wind tunnel test program at NASA Lewis Research Center verified the feasibility of this concept during the mid 1970's. A five-fold increase in stability margin was demonstrated using a YF-12 system simulation. Considerable research lies ahead, however, to adequately address this important issue.

MIXED COMPRESSION INLET INSTABILITY IMPROVEMENT



NOZZLE PERFORMANCE

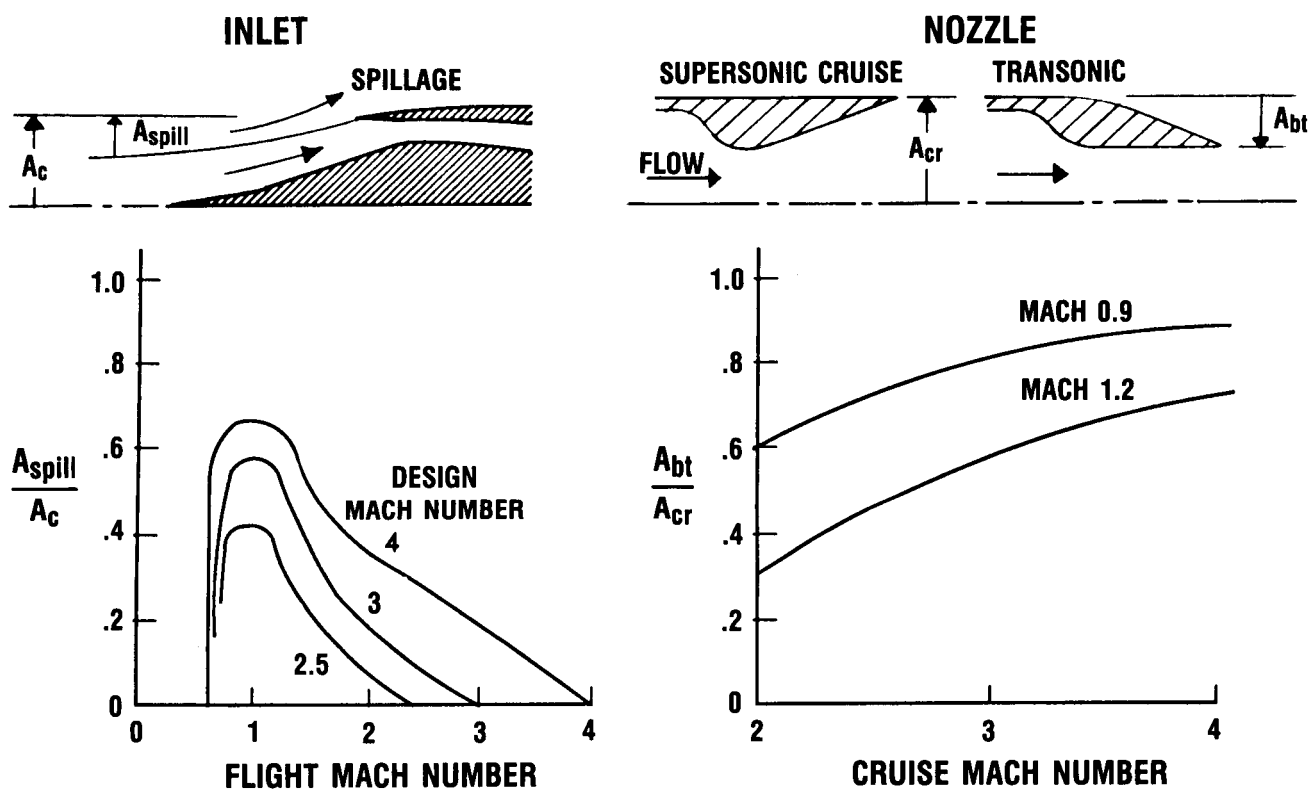
The exhaust nozzle for an SST must perform well at three critical flight conditions -- takeoff, subsonic cruise, and supersonic cruise. These experimental model test results (Lewis Research Center, 8- by 6-ft wind tunnel) of an ejector nozzle show that, while good takeoff and cruise performance was achieved, the subsonic cruise performance was disappointingly low because of flow separation over the inlet doors of the ejector. This shortfall is important because it significantly increases the reserve fuel allowance required to reach an alternate airport -- and, for long-range SST's, the amount of reserve fuel is quite large. In addition, it is critical to obtain high nozzle performance at the transonic thrust minus drag "pinch point" to minimize inlet-engine flow matching penalties.



TRANSONIC PROPULSION SYSTEM DRAG

Just as exhaust nozzle performance is critical during transonic flight, so also is the minimization of transonic installation losses associated with inlets and nozzles. The transonic inlet spillage drag, for example, can exceed the entire airframe drag for high design Mach numbers. This problem arises from a major mismatch in inlet flow-swallowing capacity (too much) compared to the engine demand. Likewise, the nozzle boattail drag penalty rises rapidly with design cruise speed. Finding solutions to these installation problems is absolutely essential to achieve an acceptable airplane design.

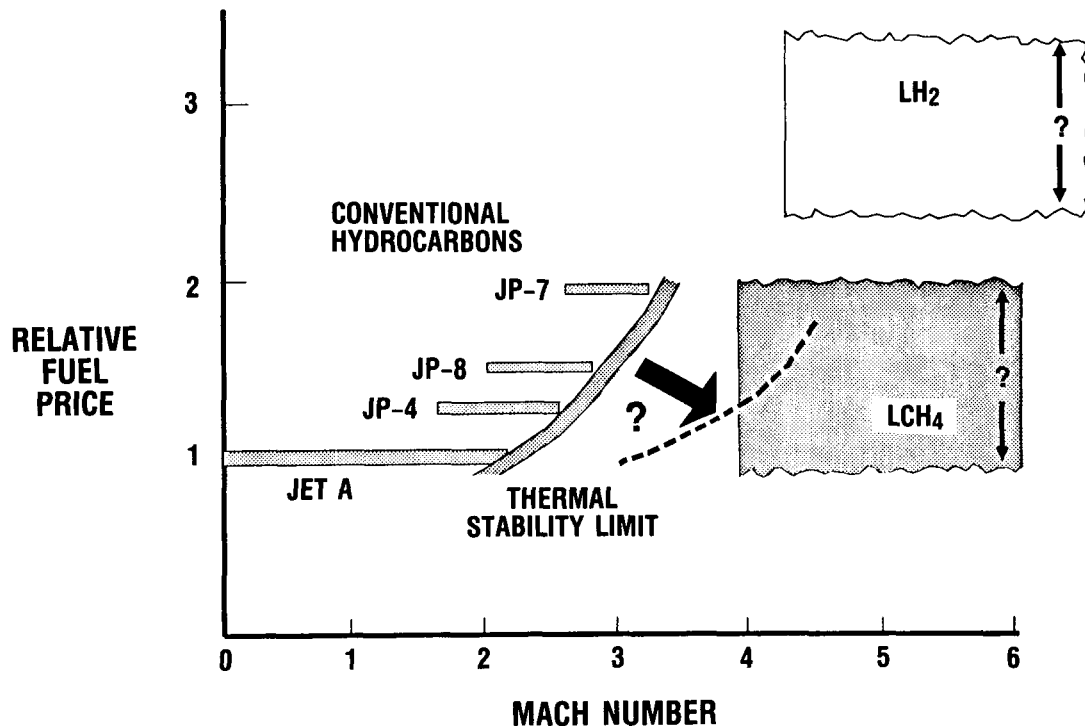
TRANSONIC PROPULSION SYSTEM DRAG



THE HIGH-SPEED TRANSPORT FUEL ISSUE

Conventional jet fuels cannot withstand the high temperatures associated with flight speeds in excess of about Mach 2. If subjected to temperatures above approximately 250 °C (time dependent also), they thermally decompose and form coke deposits that clog fuel-supply components. Consequently, a challenge exists to extend the thermal stability of conventional jet fuel (Jet A) to higher temperatures without incurring a significant fuel price increase -- either in the fuel manufacture or associated with special fuel transportation and handling requirements (such as with JP-7 and cryogenics). While the practical use of hydrogen lies far into the future, liquid methane or LNG remains as an intriguing possibility because of its current low price and high thermal stability. Endothermic fuels offer more heat sink capacity, but are fraught with offsetting practical and economic penalties. Uncertain future fuel prices and infrastructure costs cloud the issue of fuel selection and, consequently, airplane design speed as well.

THE HIGH-SPEED TRANSPORT FUELS ISSUE



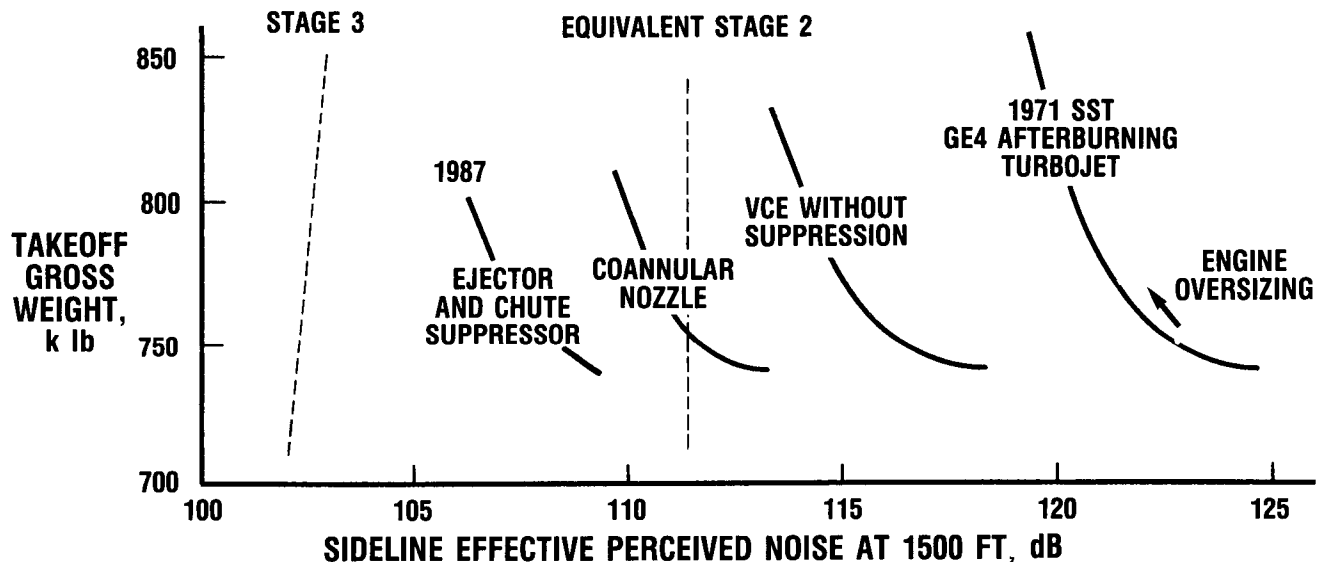
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PROGRESS IN SST TAKEOFF-NOISE REDUCTION

The first generation of hypothetical U.S. SST's of the early 1970's used afterburning turbojets and would have provoked the irritation of many people living around major airports. Reducing their high jet exhaust velocities (over 4000 ft/s) by oversizing the engines and throttling back during takeoff reduces noise somewhat, but it also increases airplane size too rapidly to be an effective method for more than a few dB. Each curve represents a series of various amounts of engine oversizing for a fixed mission. Considerable noise reduction progress evolved during the 1970's through a combination of variable-cycle features and many noise suppression concepts experimentally tested. However, even this progress is insufficient to meet current FAR 36 Stage III requirements. Much research lies ahead if we are to achieve a quiet SST without excessive noise reduction penalties.

PROGRESS IN SST TAKEOFF NOISE REDUCTION

MACH 2.4 TO 3.2 EXPERIMENTAL DATA BASE



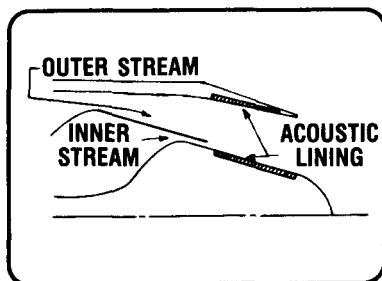
CONCLUSION: CONSIDERABLE RESEARCH EFFORT WARRANTED

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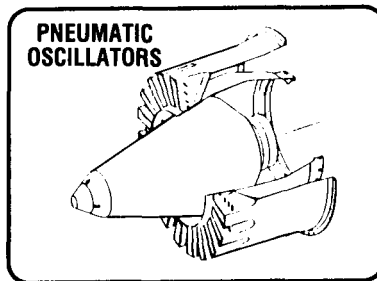
JET NOISE REDUCTION CONCEPTS

Some of the noise reduction concepts illustrated here have been explored in axisymmetric configurations suitable for Mach 2-3 airplanes. These concepts need data base extensions for two-dimensional nozzles suitable for higher flight speeds. Other concepts have practically no data base at all and are quite speculative. For example, the concept of cancelling source noise by superimposing an out of phase second source has made significant strides recently and appears suitable for discrete frequency noises such as produced by a propeller. Extending this idea to cancel broadband jet noise with passive secondary noise sources (pneumatic oscillators) represents a very speculative and technically challenging strategy. The remote augmented thrust system concept guarantees low noise with its high mass flow, low pressure ratio fan. But it introduces different problems -- notably, how to integrate the remote deployable takeoff fans into the airframe.

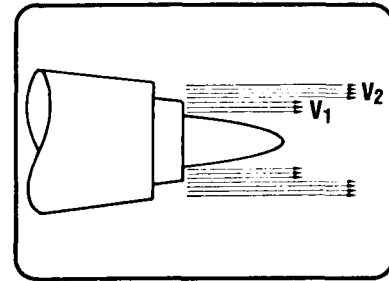
JET NOISE REDUCTION CONCEPTS



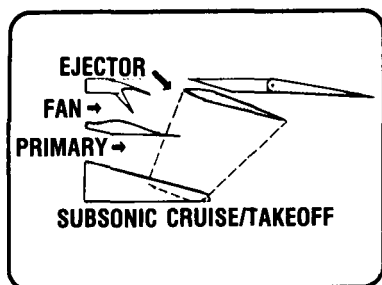
ACOUSTIC LINING



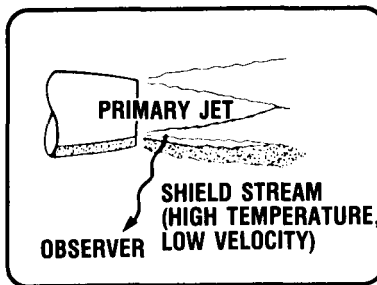
SUPPRESSOR



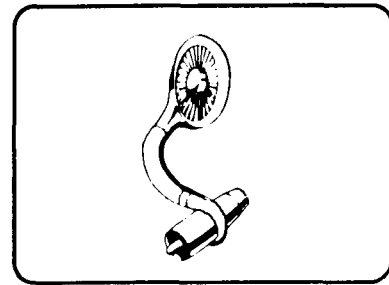
INVERTED VELOCITY PROFILE



EJECTOR



THERMAL ACOUSTIC SHIELD



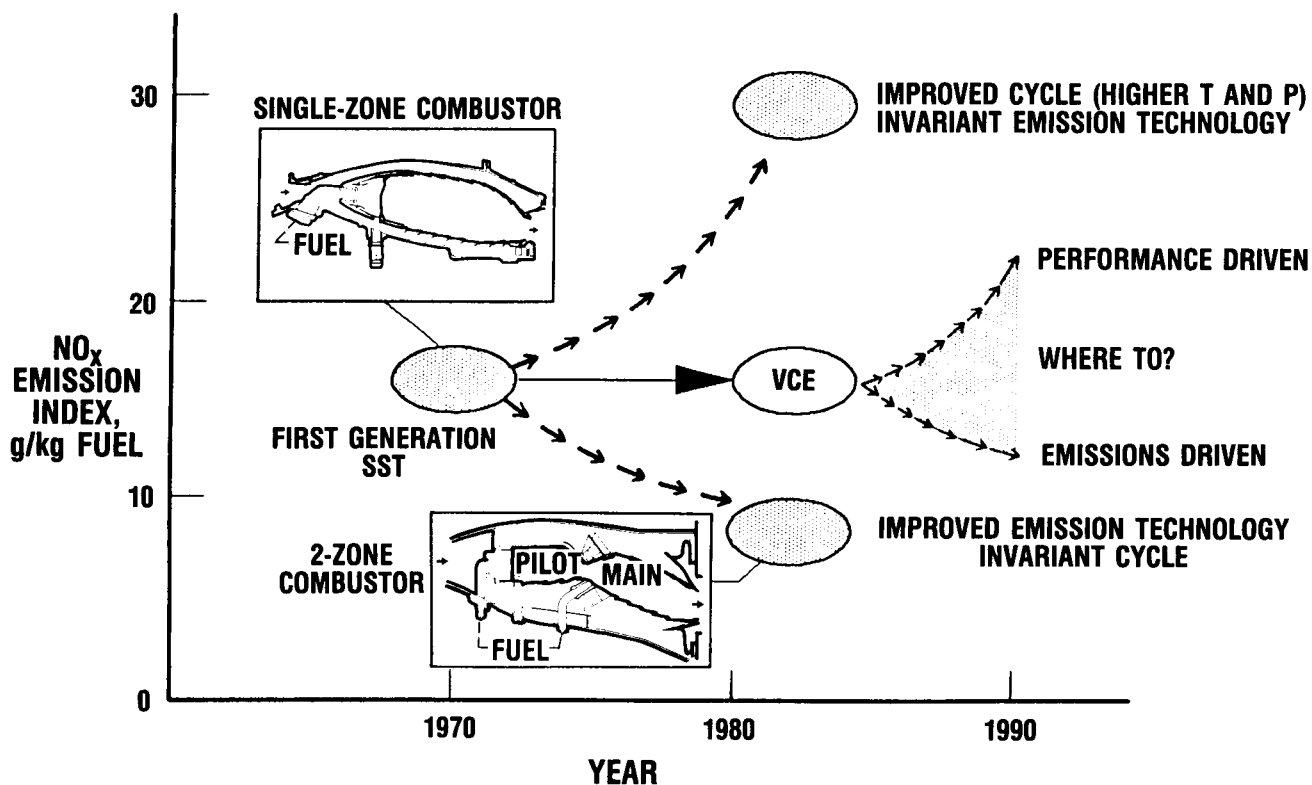
REMOTE AUGMENTED THRUST SYSTEM

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PROGRESS IN SST CRUISE NO_x-EMISSION REDUCTION

Presently, it does not appear that we have a known problem with SST engine emissions. There is some concern, however, that we might have a future problem if ongoing analyses conclude that significant upper atmospheric ozone depletion would be caused by a fleet of NO_x-emitting SST's. Previous airport pollution concerns precipitated a NASA emissions-reduction research program that led to the development of several control mechanisms including two-zone combustors. The 1970's engines had single-zone combustors that had their high-power efficiency compromised to obtain good low-power ignition and stability. The improved two-zone combustors used a pilot stage optimized for idle conditions and a main stage optimized for cruise power. This resulted in leaner, well-mixed cruise combustion with approximately one-half as much cruise NO_x emission assuming the engine cycle remains unchanged. However, our continued quest for higher overall engine efficiency produces ever higher cycle temperatures which increases NO_x production. Hence, the final engine designs of the supersonic cruise research (SCR)/VCE program, if built, would have produced about as much NO_x as the actual engines introduced a decade earlier. Today, we face the same dilemma: performance-driven designs will increase NO_x, while emissions-driven designs will reduce performance.

PROGRESS IN SST CRUISE NO_x EMISSION REDUCTION

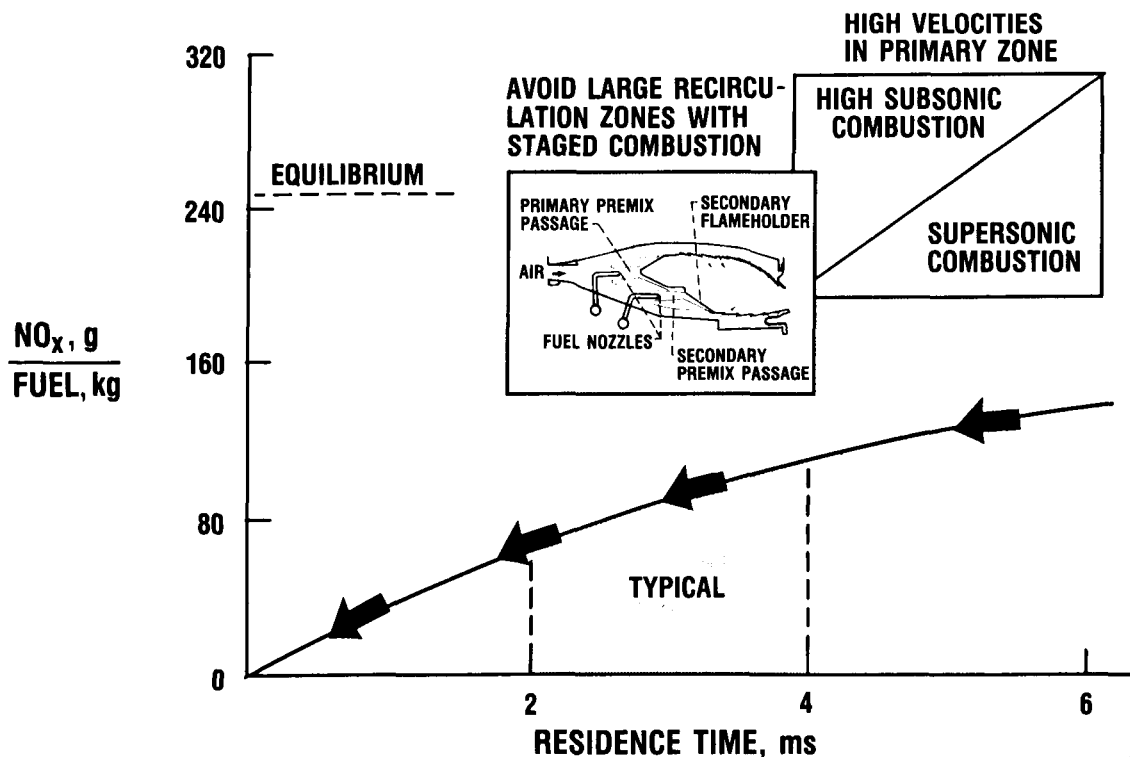


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NO_x-EMISSIONS-REDUCTION CONCEPTS

One approach to reduce NO_x emissions is to reduce the flame temperature. Another approach is to reduce the residence time of the combustion gas at high temperatures. In the latter approach, two concepts worth pursuing are (1) increasing the velocity through the combustor, and (2) avoiding large recirculation regions within the primary combustion zone. Increasing the combustion velocity to relatively high-subsonic values involves finding means to avoid excessive pressure losses, as well as maintaining good combustion stability and ignition characteristics. Avoiding large pockets of recirculating hot gases in the primary zone also reduces stability characteristics and, thereby, requires the implementation of other stability-enhancing features.

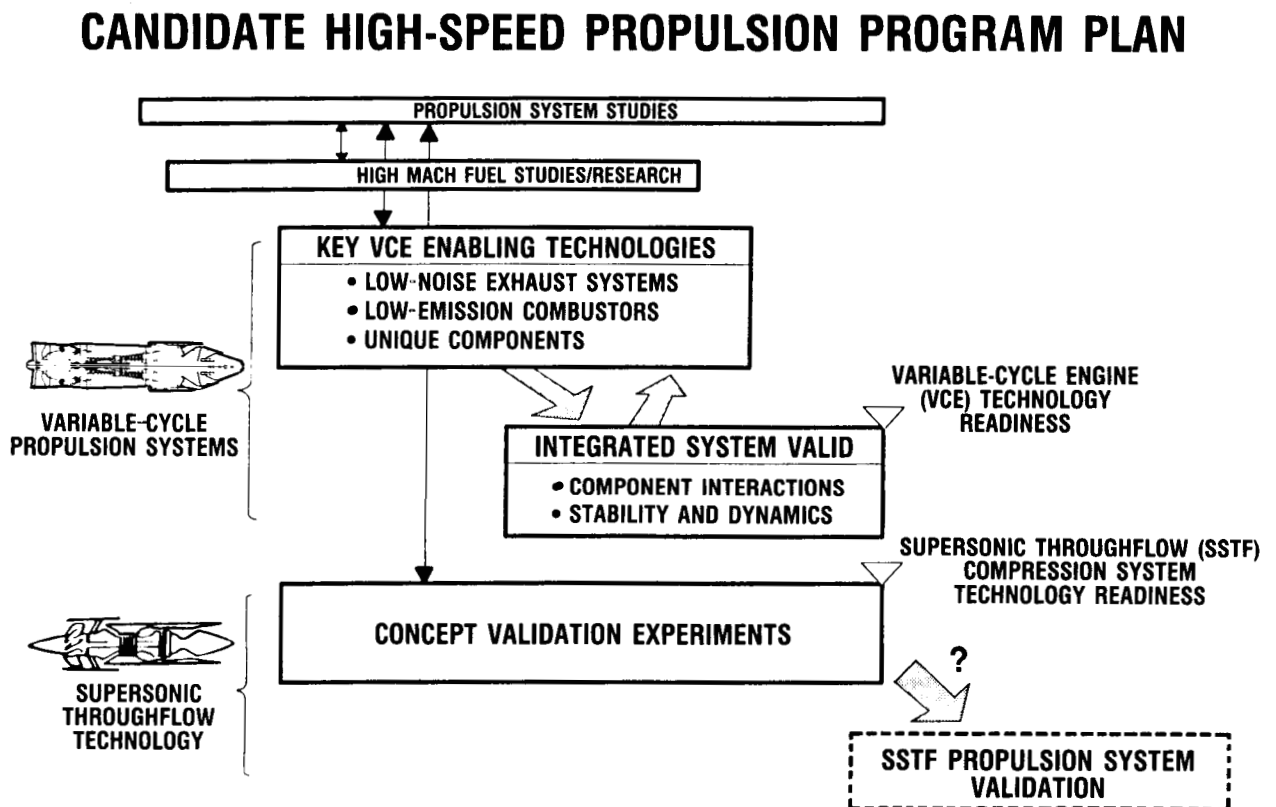
NO_x-EMISSIONS-REDUCTION CONCEPTS REDUCE RESIDENCE TIME AT HIGH TEMPERATURE



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CANDIDATE HIGH-SPEED PROPULSION PROGRAM PLAN

As the 21st century approaches, it is becoming increasingly clear that efficient supersonic cruise flight is within our technological reach. Many challenging propulsion problems need to be addressed, however, before a state of technology readiness is achieved. One possible program plan entails a two-pronged approach: a near-term effort aimed at variable-cycle engine concepts incorporating very aggressive discipline and component technologies, and a far-term effort focused on validating supersonic throughflow technology which offers even higher potential benefits. Continued propulsion system studies as well as a high-speed fuel and fuel systems effort are also needed. Attainment of the propulsion goals outlined herein would indeed revolutionize aircraft capability for the future.



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SUPERSONIC THROUGHFLOW FANS FOR HIGH-SPEED AIRCRAFT

Calvin L. Ball

ABSTRACT

Increased need for more efficient long-range supersonic flight has revived interest in the supersonic throughflow fan as a possible component for advanced high-speed propulsion systems. A fan that can operate with supersonic inlet axial Mach numbers would reduce the inlet losses incurred in diffusing the flow from supersonic Mach numbers to a subsonic one at the fan face. In addition, the size and weight of an all-supersonic inlet will be substantially lower than those of a conventional inlet. However, the data base for components of this type is practically nonexistent. Therefore, in order to furnish the required information for assessing the potential for this type of fan, the NASA Lewis Research Center has begun a program to design, analyze, build, and test a fan stage that is capable of operating with supersonic axial velocities from inlet to exit. The objectives are to demonstrate the feasibility and potential of supersonic throughflow fans, to gain a fundamental understanding of the flow physics associated with such systems, and to develop an experimental data base for design and analysis code validation.

This presentation provides a brief overview of past supersonic throughflow fan activities; discusses technology needs; describes the design of a supersonic throughflow fan stage, a facility inlet, and a downstream diffuser; and presents the results from the analysis codes used in executing the design. Also presented is a unique engine concept intended to permit establishing supersonic throughflow within the fan on the runway and maintaining the supersonic throughflow condition within the fan throughout the flight envelope.

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SUPERSONIC THROUGHFLOW FANS FOR HIGH-SPEED AIRCRAFT

The NASA Lewis Research Center has embarked on a program to develop the technology for supersonic throughflow fans applicable to high-speed aircraft propulsion systems. We feel this technology could revolutionize high-speed aircraft design and performance. The payoffs of this technology were quantified in the previous presentation.

OUTLINE

This presentation provides a brief overview of past supersonic throughflow activities; discusses technology needs; describes the design and analysis of a supersonic throughflow fan stage, a facility inlet, and a downstream diffuser; and presents a unique engine concept incorporating a supersonic throughflow fan.

OUTLINE

- **BACKGROUND**
- **TECHNOLOGY NEEDS AND STATUS**
- **NASA STF FAN PROGRAM**
- **UNIQUE ENGINE CONCEPT**
- **SUMMARY**

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BACKGROUND

Ferri, in 1956, was the first to point out the potential advantages of supersonic inflow compression systems. In 1961, Savage, Boxer, and Erwin studied the starting characteristics in transitioning to supersonic inflow. Under Air Force sponsorship in 1967, General Applied Science Laboratory (GASL), with Detroit Diesel Allison (DDA) as a subcontractor, and United Technologies Research Center (UTRC) conducted design studies and proposed turbojet engine concepts incorporating supersonic throughflow compressors. Also in 1967, Boxer proposed a high-bypass-ratio turbofan engine/ramjet combination with a variable-pitch supersonic inflow compressor. In 1975, Breugelmans conducted the most thorough supersonic throughflow fan experiment to date. In 1978, Franciscus presented the results of his first analysis showing significant payoffs of supersonic throughflow fan engines for supersonic cruise aircraft.

BACKGROUND

- FERRI WAS FIRST TO PROPOSE SUPERSONIC INFLOW COMPRESSORS (1956)
- SAVAGE, BOXER, AND ERWIN STUDIED STARTING CHARACTERISTICS (1961)
- GASL/DDA AND UTRC PROPOSED TURBOJET ENGINE CONCEPTS (1967)
- BOXER PROPOSED A TURBOFAN/RAMJET COMBINATION (1967)
- BREUGELMANS CONDUCTED MOST THOROUGH EXPERIMENT TO DATE (1975)
- FRANCISCUS PRESENTED THE RESULTS OF HIS FIRST STUDY (1978)

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TECHNOLOGY NEEDS

A technology need is to extend and validate the codes. In moving into the supersonic flow regime, where the data base is essentially nonexistent, applying computational methods in the design process should greatly enhance the quality of the experiment. Experiments are needed to obtain data for flow physics modeling and code validation and to demonstrate subsonic performance, transition, and supersonic performance.

TECHNOLOGY NEEDS

- **EXTEND AND VALIDATE CODES**
 - **SUPERSONIC THROUGHFLOWS**
 - **ENDWALL BOUNDARY LAYER FLOWS**
 - **BLADE ROW INTERACTIONS**
 - **UNSTEADY FLOWS**
 - **DESIGN AND OFF-DESIGN PERFORMANCE PREDICTIONS**
- **CONDUCT EXPERIMENTS TO OBTAIN DATA FOR FLOW PHYSICS MODELING AND CODE VERIFICATION AND TO DEMONSTRATE PERFORMANCE**
 - **SUBSONIC PERFORMANCE**
 - **TRANSITION**
 - **SUPERSONIC PERFORMANCE**
 - **CHOKE, STALL, AND UNSTART**
 - **DISTORTION TOLERANCE**
 - **FLUTTER AND FORCED RESPONSE**

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CODE STATUS

The indicated codes have been modified to accommodate supersonic throughflow velocities. However, because of the lack of data, the codes, with the exception of the three-dimensional parabolized Navier-Stokes code, have not been validated. This code has been validated for duct flows. All of the codes were used in the design and analysis of the NASA supersonic throughflow fan experiment. They will be validated as experiment results become available.

CODE STATUS

THE FOLLOWING DESIGN AND ANALYSIS CODES HAVE BEEN MODIFIED AND APPLIED, BUT NOT YET VALIDATED FOR THE SUPERSONIC FLOW REGIME:

- **AXISYMMETRIC DESIGN CODE (CROUSE)**
- **1-D STAGE STACKING CODE (STEINKE)**
- **AXISYMMETRIC OFF-DESIGN CODE (CROUSE)**
- **QUASI-3-D THIN SHEAR LAYER NAVIER-STOKES CODE (CHIMA)**
- **3-D PASSAGE AVERAGE STAGE CODE (ADAMCZYK)**
- **SUPERSONIC THROUGHFLOW FLUTTER CODE (RAMSEY)**
- **3-D EULER CODE WITH 2-D BOUNDARY LAYER MODEL (DENTON)**
- **3-D PARABOLIZED NAVIER-STOKES CODE (PNS)***
- **3-D UNSTEADY EULER CODE (WHITFIELD)**

***VALIDATED FOR DUCT FLOW**

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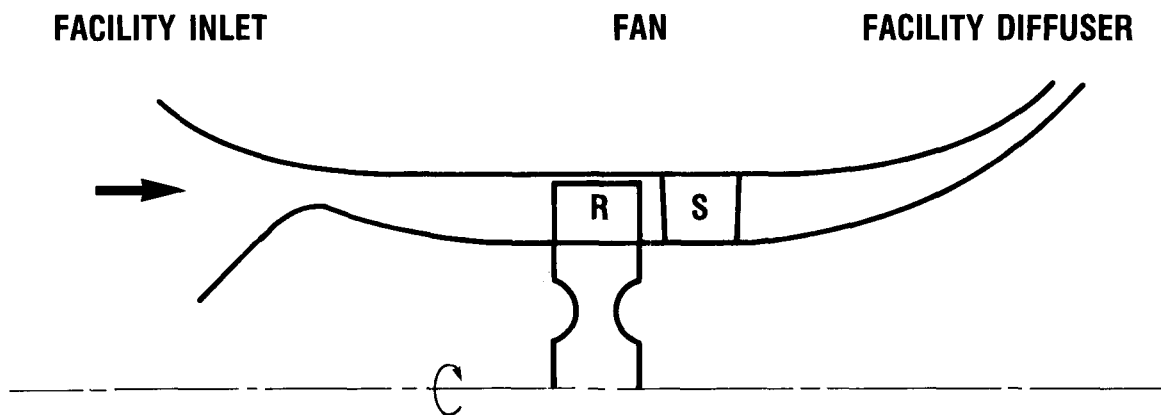
NASA SUPERSONIC THROUGHFLOW FAN PROGRAM

In discussing the NASA supersonic throughflow fan design, particular attention will be given to the results obtained from the analysis codes and to how they were used to guide the design. The detailed flow physics gleaned from the codes will be highlighted.

SUPERSONIC THROUGHFLOW FAN

This figure depicts a supersonic throughflow fan, the facility inlet needed to accelerate the flow to supersonic velocities at the fan face, and the diffuser needed downstream to decelerate the supersonic flow leaving the fan to subsonic conditions downstream. The design fan-face Mach number is 2.0 and the exit Mach number is 2.9. The fan was designed with a constant annulus area. The design pressure ratio and tip speed were selected to be representative of those required of a turbofan engine fan operating at supersonic cruise conditions.

SUPERSONIC THROUGHFLOW FAN



| | |
|-----------------------|-------------|
| PRESSURE RATIO | 2.45 |
| AXIAL MACH NUMBER | |
| INLET | 2.0 |
| EXIT | 2.9 |
| ROTOR TIP SPEED | 1500 ft/sec |

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DESIGN CRITERIA

This figure presents the design criteria established to guide design of the fan, the facility inlet, and the downstream diffuser.

DESIGN CRITERIA

APPLY COMPUTATIONAL TWO- AND THREE-DIMENSIONAL INVISCID AND VISCOUS CODES TO

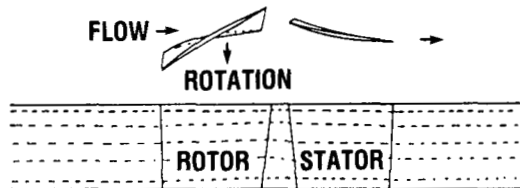
- **FAN**
 - **ENSURE STARTED CONDITION**
 - **MAINTAIN SUPERSONIC THROUGHFLOW VELOCITIES THROUGHOUT COMPRESSION SYSTEM**
 - **ENSURE THAT SHOCK STRUCTURE IS CAPTURED WITHIN BLADED PASSAGES**
 - **CONTROL SUCTION AND PRESSURE SURFACE GRADIENTS TO MINIMIZE STRENGTH OF INTERNAL COMPRESSION AND EXPANSION WAVE SYSTEM**
- **FACILITY INLET**
 - **ACHIEVE UNIFORM VELOCITY DISTRIBUTION AT FAN INLET**
 - **MINIMIZE ENDWALL BOUNDARY LAYERS ENTERING FAN**
- **FACILITY DIFFUSER**
 - **MINIMIZE DIFFUSION LOSSES THROUGH A SERIES OF CONTROLLED WEAK COMPRESSION WAVES**
 - **ENSURE STARTED CONDITION**

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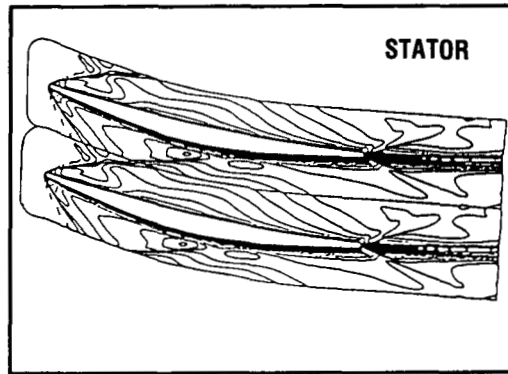
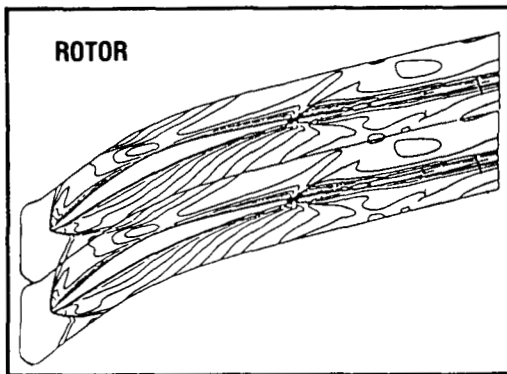
DESIGN PROCEDURE - AXISYMMETRIC DESIGN CODE AND
QUASI-3-D THIN SHEAR LAYER NAVIER-STOKES CODE

In the design of the fan an axisymmetric design code was used to obtain initial blade shapes. The quasi-three-dimensional thin shear layer Navier-Stokes code was used to analyze the design. The design was then adjusted by using the design codes, and the process was repeated until the desired loading distributions and wave patterns were achieved. The Mach number contours for the rotor and stator show that the waves off the leading edge are contained within the bladed passage. Also, the expansion waves off the suction surface tend to cancel the compression waves off the pressure-surface leading edge, thus reducing the pressure gradient along the suction surface. At the trailing edge the strength of the expansion and compression waves was minimized by controlling the loading near the trailing edge.

DESIGN PROCEDURE
AXISYMMETRIC DESIGN CODE



QUASI-3-D THIN SHEAR LAYER NAVIER-STOKES CODE
(MACH NUMBER CONTOURS)

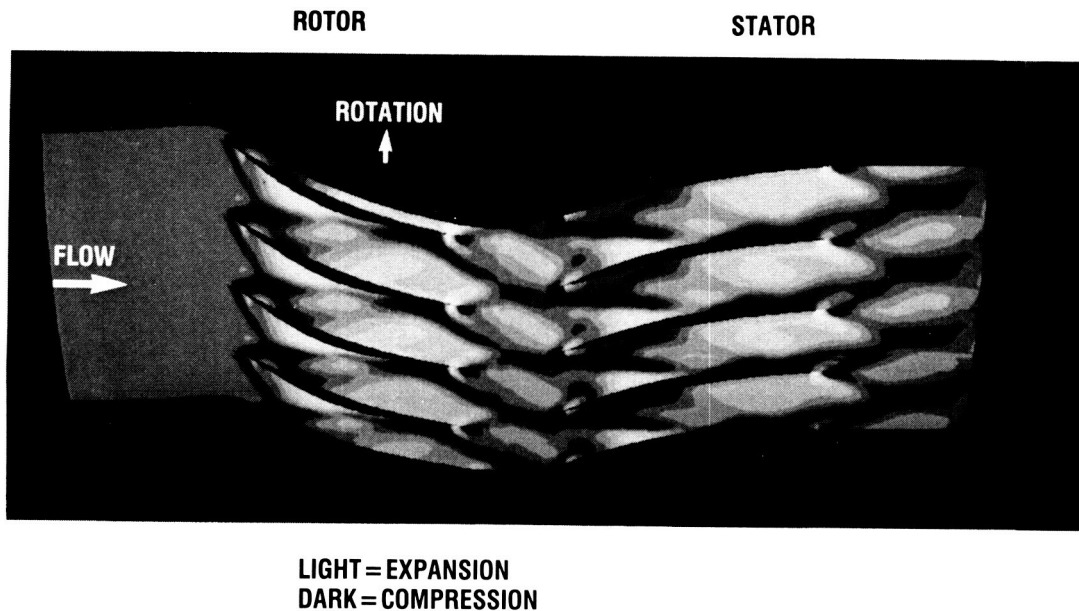


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3-D UNSTEADY EULER CODE SHOWING INTERACTIVE WAVE PATTERNS

This figure shows results obtained from a three-dimensional unsteady Euler code used to study the rotor/stator flowfield interactions with supersonic throughflow. Computer graphics were used to obtain the interactive wave patterns for a given index of the rotor relative to the stator. The picture can be thought of as a schlieren photograph with the light patterns being expansion waves and the dark patterns, compression waves. The effect of the time-dependent flowfields behind the rotor on the stator flowfield can best be seen from the next figure.

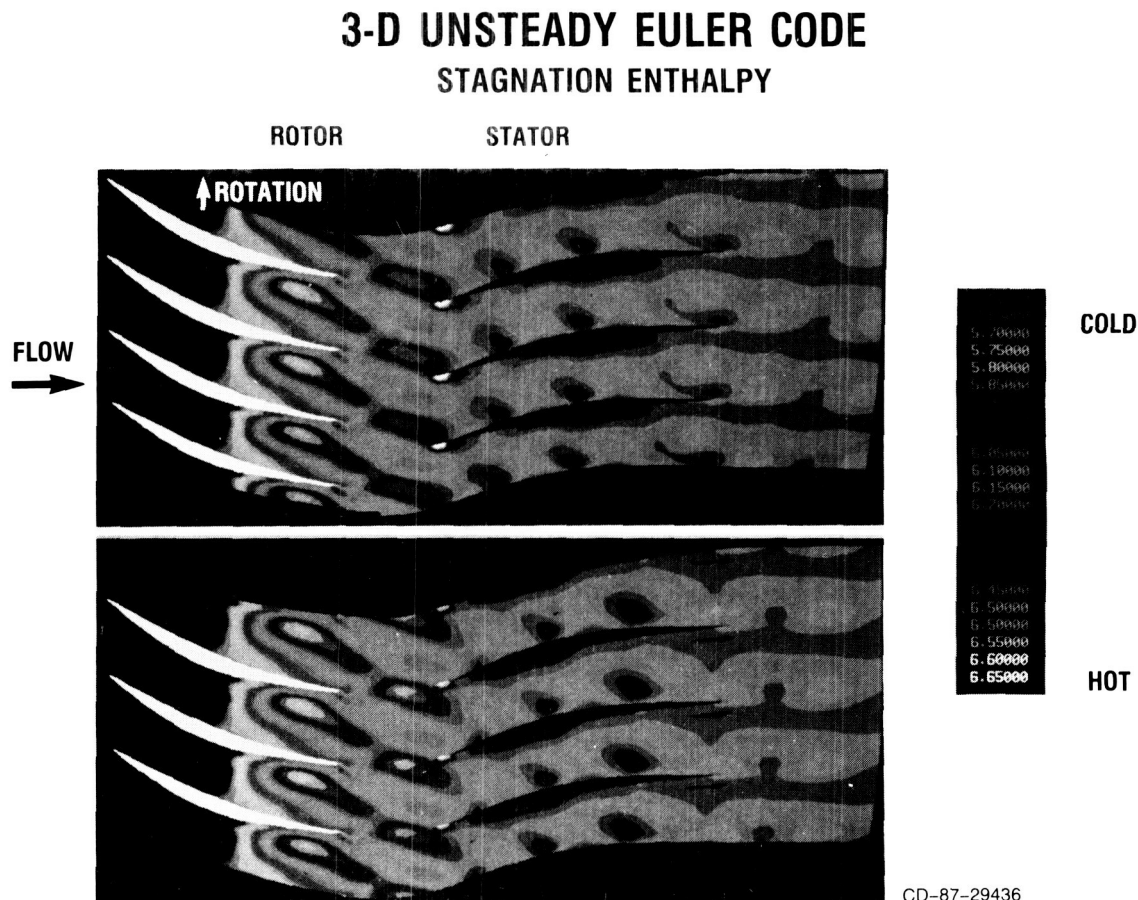
**3-D UNSTEADY EULER CODE
INTERACTIVE WAVE PATTERNS**



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3-D UNSTEADY EULER CODE SHOWING STAGNATION ENTHALPY

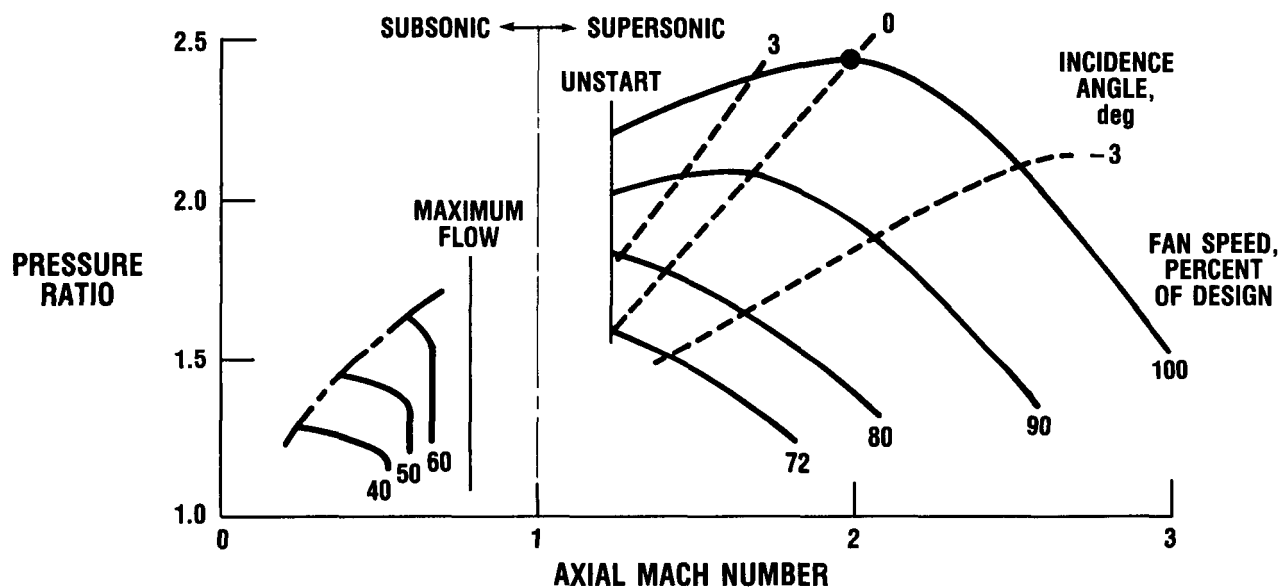
This figure shows the stagnation enthalpy, and thus temperature, for two different indexes of the rotor blades relative to the stators. The interactive wave patterns within and exiting the rotor result in a time-dependent flowfield entering the stator. This unsteady flowfield relative to the stator appears to result in cyclic movement fore and aft of the stator leading-edge compression wave, which emanates from the pressure surface. Wave motion is nonlinear, with more energy being added when the shock moves forward than is subtracted when the shock moves rearward. Further analysis is needed to fully understand this phenomenon. The cyclic nature of the local temperature is apparent from the difference in the magnitudes of the local white (highest temperature) regions. However, the unsteady aspects of the flowfield can best be seen from a motion picture generated from the three-dimensional unsteady Euler code analysis.



PREDICTED FAN PERFORMANCE MAP - PRESSURE RATIO VERSUS AXIAL MACH NUMBER

The predicted performance map for the supersonic throughflow fan was derived by using a combination of codes including the off-design axisymmetric code and the quasi-three-dimensional viscous code. Presenting the performance as a function of inlet axial Mach number results in a performance map similar to subsonic/transonic fan maps.

PREDICTED FAN PERFORMANCE MAP PRESSURE RATIO VS AXIAL MACH NUMBER

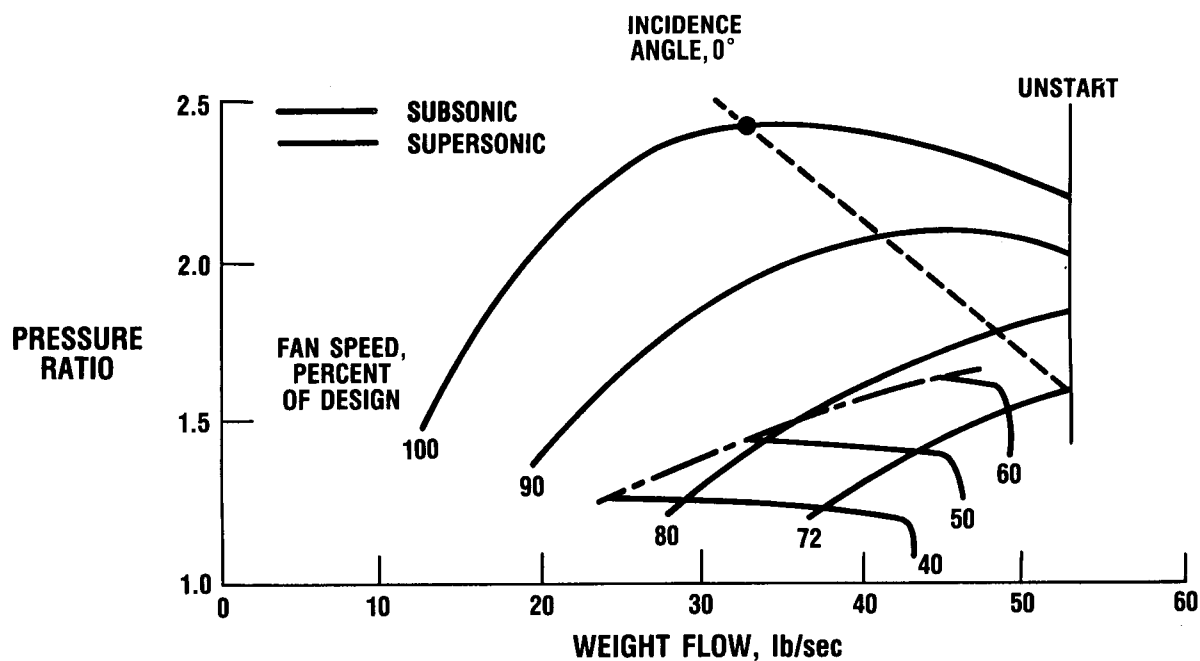


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PREDICTED FAN PERFORMANCE MAP - PRESSURE RATIO VERSUS WEIGHT FLOW

This figure reflects the reduction in flow capacity on the supersonic throughflow side of the map as the inlet axial Mach number is increased.

PREDICTED FAN PERFORMANCE MAP PRESSURE RATIO VS WEIGHT FLOW

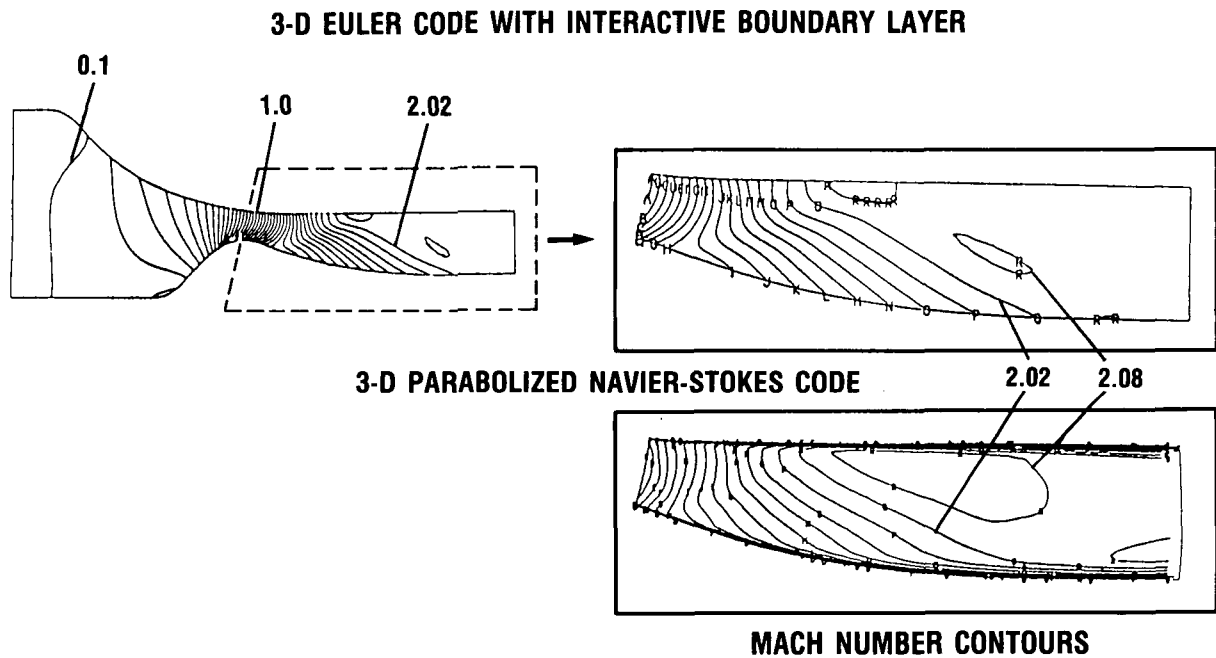


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FACILITY VARIABLE-INLET NOZZLE MACH NUMBER CONTOURS

This figure presents the results obtained from a three-dimensional Euler code with an interactive boundary layer routine and from a three-dimensional parabolized Navier-Stokes code in analyzing the flowfield of the variable-inlet nozzle. The nozzle was positioned to achieve the design axial Mach number of 2.0. Good agreement existed between the codes. The codes indicated that at the design condition the flow was radially uniform at the fan face and the wall boundary layers were relatively thin.

FACILITY VARIABLE-INLET NOZZLE

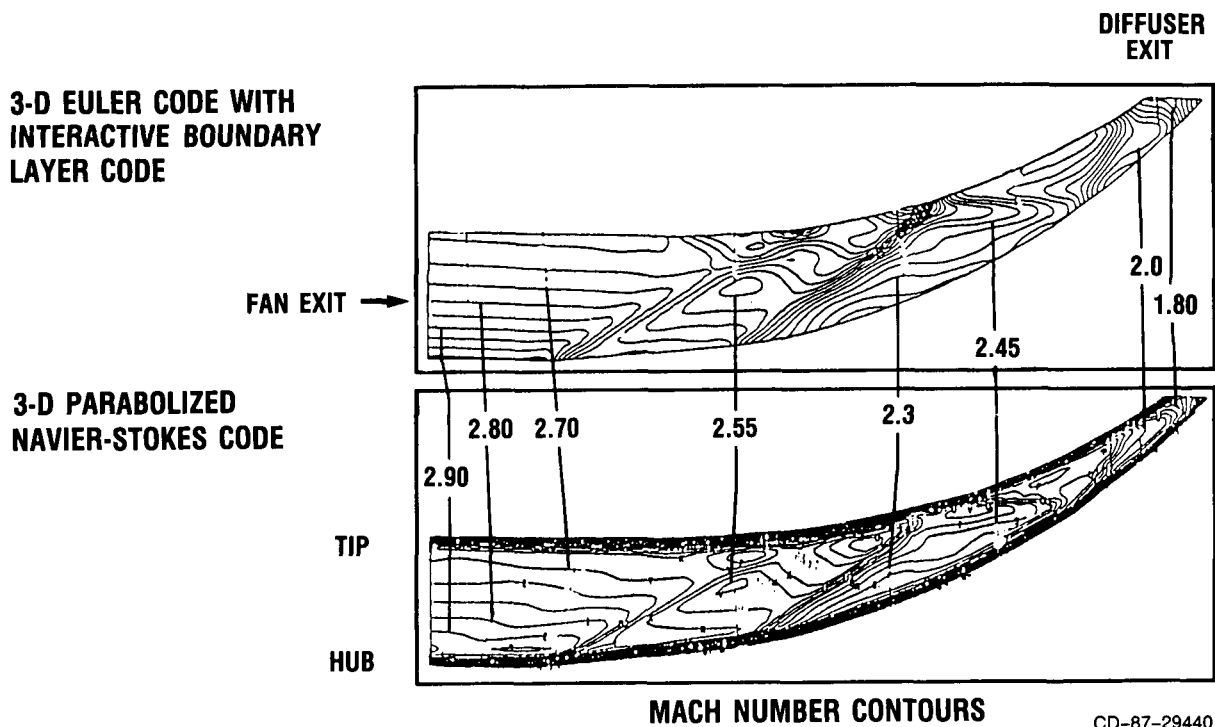


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FACILITY VARIABLE DIFFUSER MACH NUMBER CONTOURS

A similar analysis was conducted for the diffuser as for the inlet. Again, good agreement was obtained between the Euler and the viscous codes. The diffuser is designed to diffuse the flow from a fan exit Mach number of approximately 2.9 to Mach 1.8, primarily through two weak compression waves.

FACILITY VARIABLE DIFFUSER

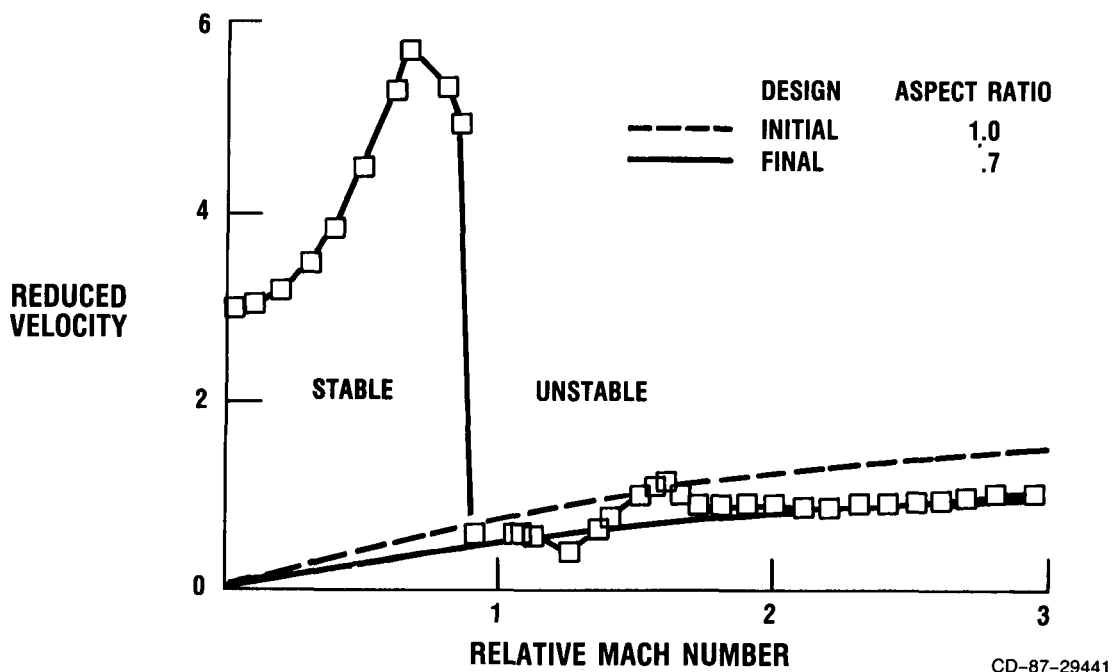


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AEROELASTIC ANALYSIS - TORSIONAL FLUTTER

This figure presents the results from an analysis of the flutter potential of the supersonic throughflow fan. Note the large reduction in stable operating range indicated by the supersonic throughflow flutter analysis at supersonic relative velocities. Even though the initially designed rotor blade was relatively low in aspect ratio, the analysis indicated a potential for supersonic torsional flutter. The design aspect ratio was further reduced in the final design to bring the rotor into the stable operating range.

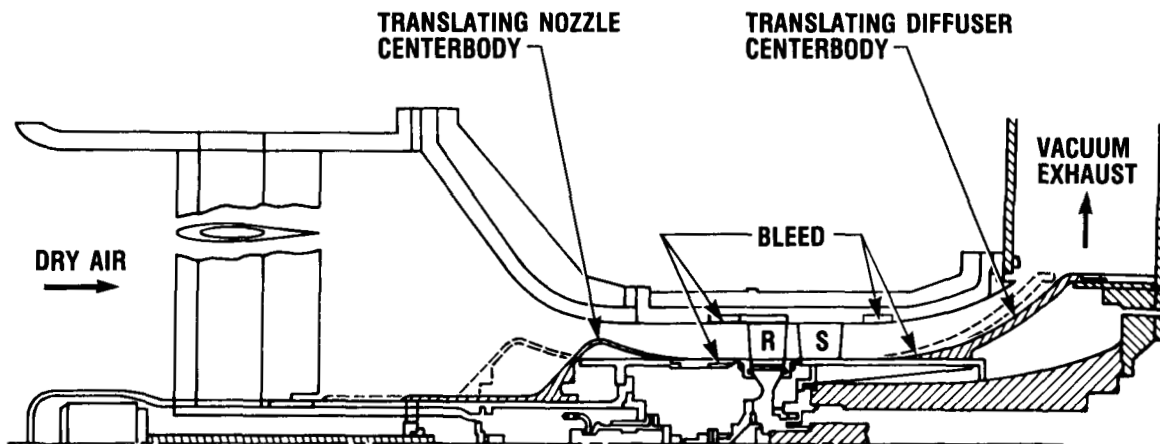
AEROELASTIC ANALYSIS OF SUPERSONIC THROUGHFLOW FAN TORSIONAL FLUTTER



TEST PACKAGE

This figure is a layout of the supersonic throughflow fan test package. The variable-inlet nozzle and the variable downstream diffuser will be used to provide control over the fan-face Mach number and the diffusion of the supersonic fan exit velocities to subsonic conditions entering the exhaust system. Boundary layer bleed capability is provided at the inlet to the fan and the diffuser.

TEST PACKAGE



PRESSURE RATIO 2.45
INLET AXIAL MACH NUMBER 2.0
ROTOR TIP SPEED 1500 ft/sec
TIP DIAMETER 20 in.

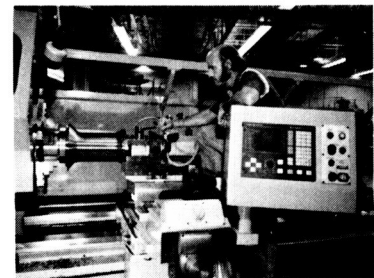
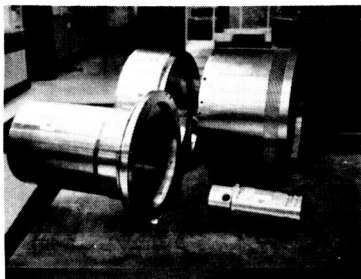
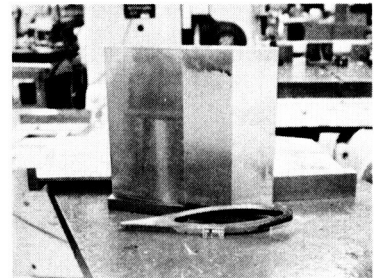
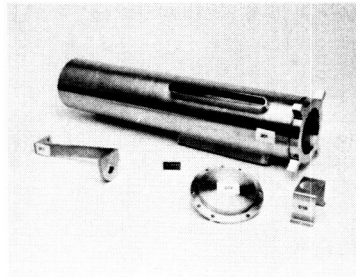
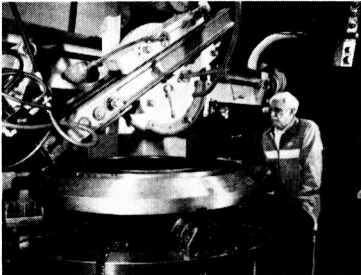
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HARDWARE FABRICATION

This figure shows some of the fan hardware in various phases of completion.

HARDWARE FABRICATION



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UNIQUE HIGH-SPEED ENGINE CONCEPT

Some of the problems raised in connection with supersonic throughflow fans for supersonic aircraft are how to "fly" such an engine system, how the fan can be made to transition to the supersonic side of the performance map and at what flight speed this would occur, how to select the design point, and unstart. The following figures present a unique engine concept that solves these problems. The transition to supersonic throughflow within the fan component is made while the airplane is on the runway. This concept's many advantages will be discussed.

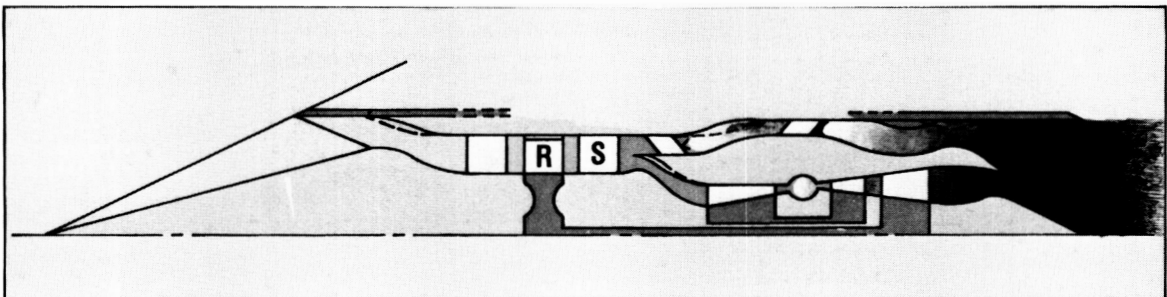
SUPERSONIC THROUGHFLOW TURBOFAN HIGH-SPEED ENGINE CONCEPT - MACH 3 DESIGN

Shown is a supersonic throughflow turbofan engine. However, the basic concept also applies to turbojet and airturboramjet cycles. The concept incorporates a short annular inlet with a variable capture and throat area to ease transitioning to supersonic throughflow within the fan on the runway and to maintain supersonic flow at the fan face throughout the flight envelope. Located downstream of the fan are annular supersonic/subsonic diffusers, one in the bypass duct and the other in the core inlet duct. The core inlet also features a variable capture area to help in flow matching and in optimizing performance. The flow entering the core compressor is diffused to subsonic conditions throughout the flight envelope. However, the duct flow is diffused subsonically for only subsonic flight Mach numbers and remains supersonic for supersonic flight. The variable-geometry features in the diffuser and nozzle are intended to achieve these goals. Duct burning may or may not be required. Some of the advantages of this concept are no forward transmission of fan noise on takeoff, ease of meeting pressure ratio requirements for takeoff and aircraft acceleration through Mach 1, potentially good subsonic cruise performance for overland operation, and, it is hoped, no variable geometry in the fan rotor.

The flowpath shown for the turbofan is consistent with the Mach 3 design. The design flight Mach number has a significant effect on the fan geometry.

SUPERSONIC THROUGHFLOW TURBOFAN HIGH-SPEED ENGINE CONCEPT

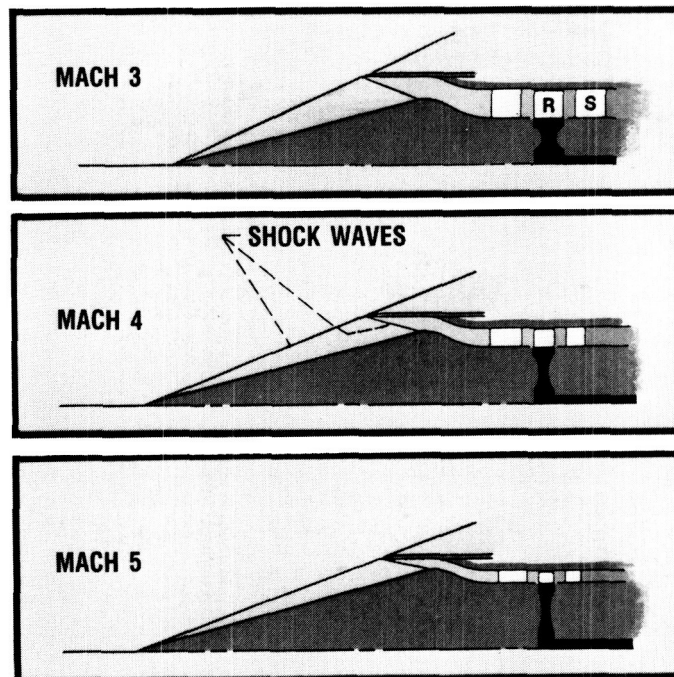
MACH 3 DESIGN



VARIOUS FLIGHT MACH NUMBER DESIGNS FOR SUPERSONIC THROUGHFLOW TURBOFAN

This figure illustrates the effect of flight Mach number on the fan geometry. For the Mach 3 condition, as shown in the previous figure, the fan hub/tip ratio is about 0.7. As the flight Mach number is increased, the passage height decreases and the hub/tip ratio increases. At Mach 5 the fan hub/tip ratio is above 0.8. The reduction in the fan tip diameter in relation to the inlet diameter is adequate to achieve the desired change in throat area with acceptable axial translation of the nozzle. By limiting the reduction in fan diameter, the gooseneck is minimized and the strength of the expansion and compression waves is reduced during supersonic operation. To achieve the low hub/tip ratio typical of transonic fans while achieving satisfactory supersonic operation would require a prohibitive gooseneck.

VARIOUS FLIGHT MACH NUMBER DESIGNS FOR SUPERSONIC THROUGHFLOW TURBOFAN

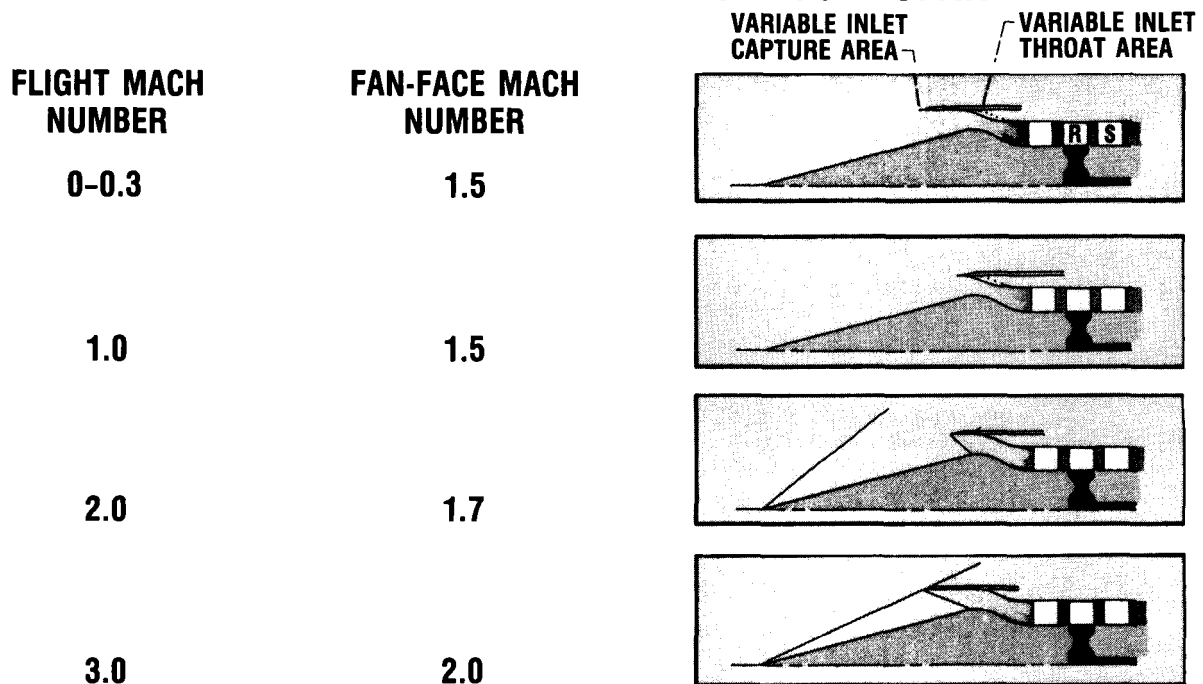


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INLET GEOMETRY CONFIGURATIONS FOR SUPERSONIC THROUGHFLOW TURBOFAN

This figure shows the inlet geometry configurations for Mach 0 to 0.3, 1.0, 2.0, and 3.0 (the assumed design point). From Mach 0 to 0.3 the inlet cowl is extended forward so that the flowpath will converge to accelerate the air to Mach 1.0 at the throat. The throat area is set relative to the fan inlet area to achieve the desired fan-face Mach number, in this case 1.5. As the flight Mach number is increased to supersonic conditions, a bow wave is formed off the inlet spike. As the Mach number becomes supersonic behind the shock, the inlet cowl is set to control the position of the internal reflected wave. At the design point of Mach 3.0 the cowl is positioned such that the leading-edge bow wave is attached to the cowl lip. The throat is opened up at Mach 2.0 and 3.0 flight to achieve the desired fan-face Mach number.

INLET GEOMETRY CONFIGURATIONS FOR SUPERSONIC THROUGHFLOW TURBOFAN

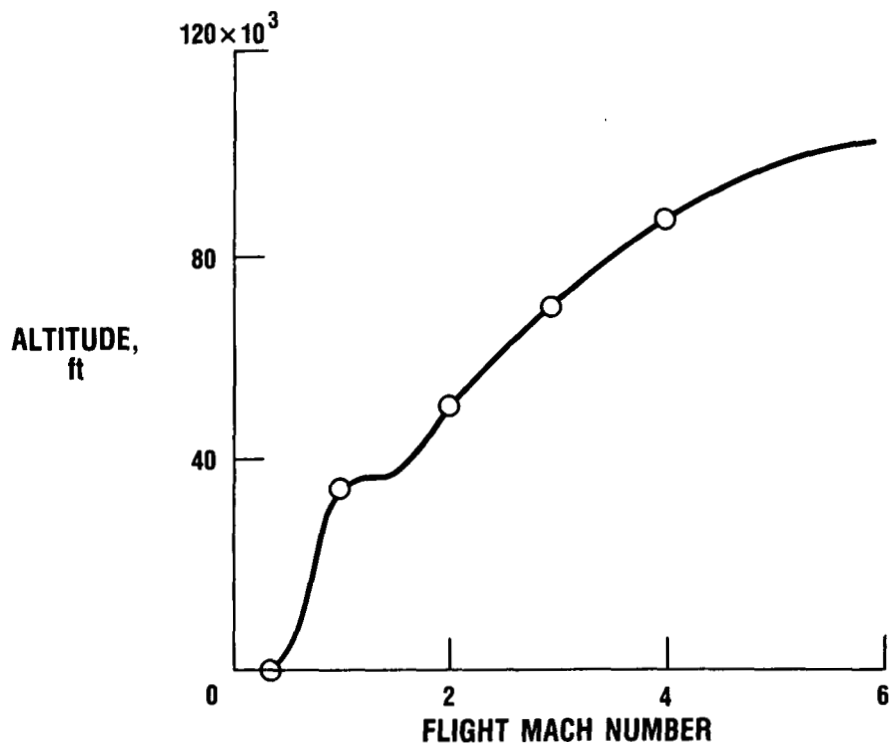


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ASSUMED HIGH-SPEED AIRCRAFT CLIMB FLIGHTPATH

This figure shows the assumed flightpath used in examining how the inlet would be configured over the flight envelope. Mach 0.3 was assumed for takeoff, transition to supersonic flight Mach numbers at 35 000 ft, and Mach 3.0 cruise at 70 000 ft.

ASSUMED HIGH-SPEED AIRCRAFT CLIMB FLIGHTPATH

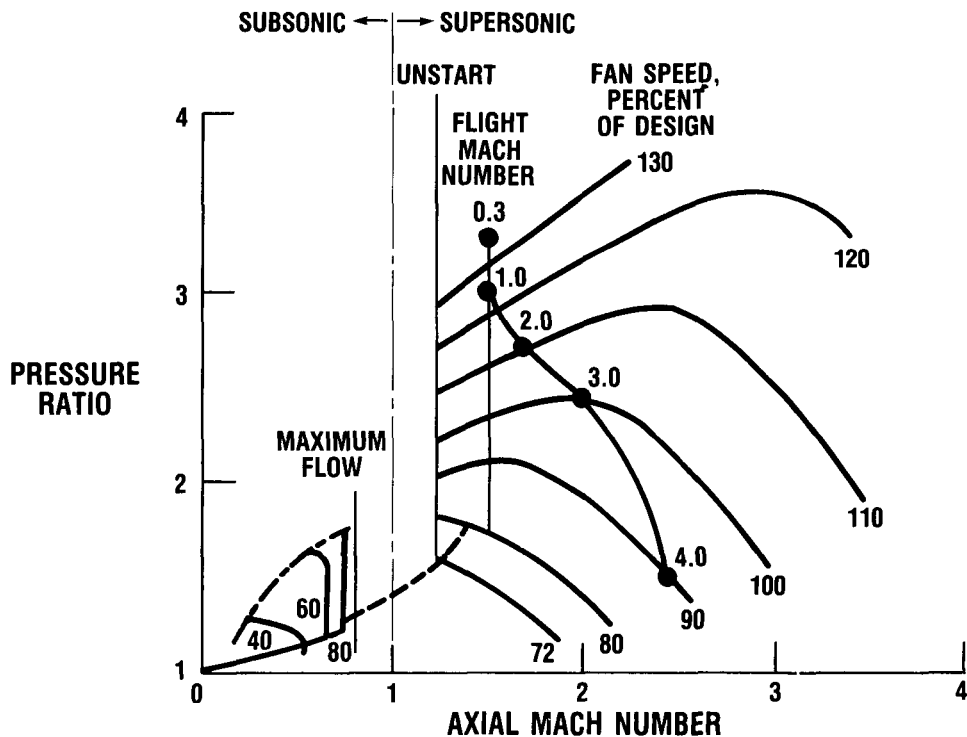


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SUPERSONIC THROUGHFLOW TURBOFAN FAN OPERATING LINE

This fan performance map shows the startup, transition, and flight operating lines. The design point was assumed to be Mach 3.0 cruise with a fan-face Mach number of 2.0 and a pressure ratio of approximately 2.5, consistent with that derived from mission analysis studies conducted by Franciscus for a Mach 3.0 transport aircraft. These same studies indicated fan pressure ratio requirements of approximately 3.3 and 3.0 for takeoff and aircraft transition to supersonic flight Mach numbers, respectively. The inlet is set to the lower design fan-face Mach number during takeoff and transition to maximize the flow and to minimize inlet bleed requirements. Predicted maximum subsonic flow before startup is shown along with the predicted unstart boundary on the supersonic side of the performance map. Note the large supersonic flow range. The startup method is to increase speed to approximately 80 percent and then close the inlet nozzle slightly. In so doing, it is predicted that the normal shock will transition through the fan. The concept requires a low load line to keep the fan out of stall during subsonic operation, even though the fan incidence will be high just prior to transitioning. As the normal shock passes through the fan, the operating point will jump to the supersonic side along the 80 percent speed line. The inlet will be adjusted to achieve the desired fan-face Mach number, and the speed will then be increased to takeoff conditions. The reverse procedure would be employed during landing.

SUPERSONIC THROUGHFLOW TURBOFAN FAN OPERATING LINE



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SUMMARY

In summary, mission studies conducted by Franciscus have shown significant benefits from supersonic throughflow fans. The design and analysis conducted on the NASA supersonic throughflow fan shows promise for such a stage. However, an experiment is strongly needed to demonstrate transition and to validate the computational codes. Off-design analysis is continuing with emphasis on the use of Chima's quasi-three-dimensional viscous code. Rotor/stator interactions will be investigated at off-design conditions by using Whitfield's code. The fan is now in fabrication, and testing is scheduled for the end of 1989.

SUMMARY

- OFF-DESIGN PERFORMANCE ANALYSIS CONTINUING
 - SUBSONIC/TRANSITION
 - SUPERSONIC
- UNSTEADY AERODYNAMIC ANALYSIS CONTINUING
 - ROTOR/STATOR INTERACTION
- STF FAN, FACILITY INLET, AND DOWNSTREAM DIFFUSER IN FABRICATION
- TESTING PROJECTED FOR END OF 1988

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HIGH-SPEED INLET RESEARCH PROGRAM AND SUPPORTING ANALYSES

Robert E. Coltrin

ABSTRACT

A Mach 5 cruise aircraft was studied in a joint program effort by NASA Lewis, NASA Langley, and Lockheed. The propulsion system chosen for this aircraft was an over-under turbojet/ramjet system. The ramjet portion of the inlet is to be tested in NASA Lewis' 10 X 10 SWT. To test the Mach 5.0 design in this facility, which has a maximum Mach number capability of 3.5, the 1/3-scale inlet model is to be mounted under a large plate at a negative angle of attack. By this means, the Mach 3.5 freestream flow is expanded up to the desired speed on the inlet first ramp. Goals of the test program are to obtain performance data and bleed requirements, and also to obtain analysis code validation data. The inlet was designed by using a conventional method of characteristics approach with boundary layer correction.

Supporting analysis of the inlet using a three-dimensional parabolized Navier-Stokes code (PEPSIS) indicates that sidewall shock/boundary layer interactions cause large separated regions in the corners underneath the cowl. Such separations generally lead to inlet unstart, and are thus a major concern. This sidewall glancing shock/boundary layer interaction has been previously documented with fundamental research in the Lewis 1 X 1 SWT. As a result of the analysis, additional bleed regions have been added to the inlet model sidewalls and cowl to control separations in the corners. A two-dimensional analysis incorporating bleed on the ramp is also presented.

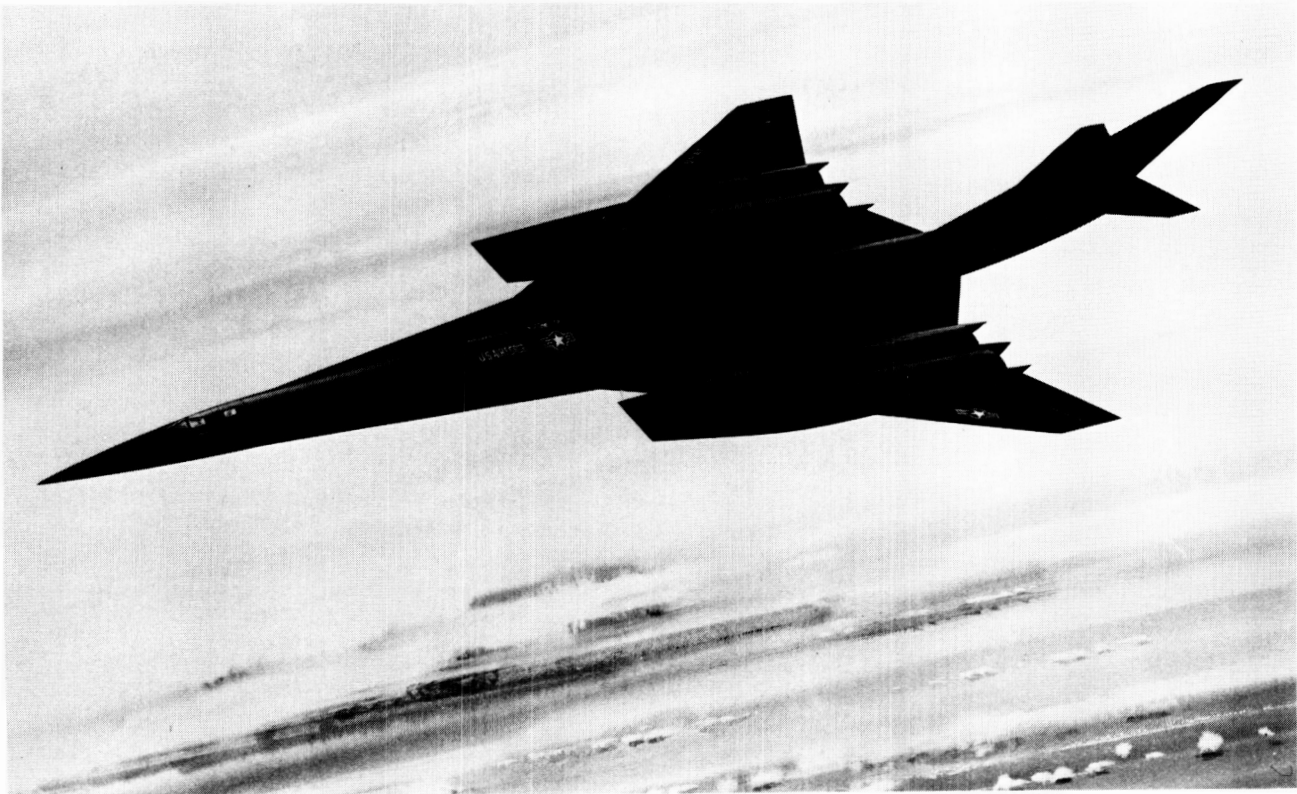
Supporting experiments for the Mach 5 program have been conducted in Lewis' 1 X 1 SWT. A small-scale model representing the inlet geometry up to the ramp shoulder and cowl lip was tested to verify the accelerator plate test technique and to obtain data on flow migration in the ramp and sidewall boundary layers. Another study explored several ramp bleed configurations to control boundary layer separations in that region.

Design of a two-dimensional Mach 5 cruise inlet represents several major challenges including multimode operation and dual flow, high temperatures, and three-dimensional airflow effects.

MACH 5 CRUISE AIRCRAFT STUDY

In 1980, a joint NASA Lewis, NASA Langley, and Lockheed California (with P&W as subcontractor) program was initiated. The purpose of this study was to define an aircraft capable of sustained high-speed cruise, and specifically, to define the propulsion system for this aircraft. The final configuration from this study is shown in the figure. The aircraft would employ four propulsion modules (two under each wing). The propulsion system chosen for this aircraft is an over-under turbojet plus ramjet system with a two-dimensional dual flow inlet and nozzle.

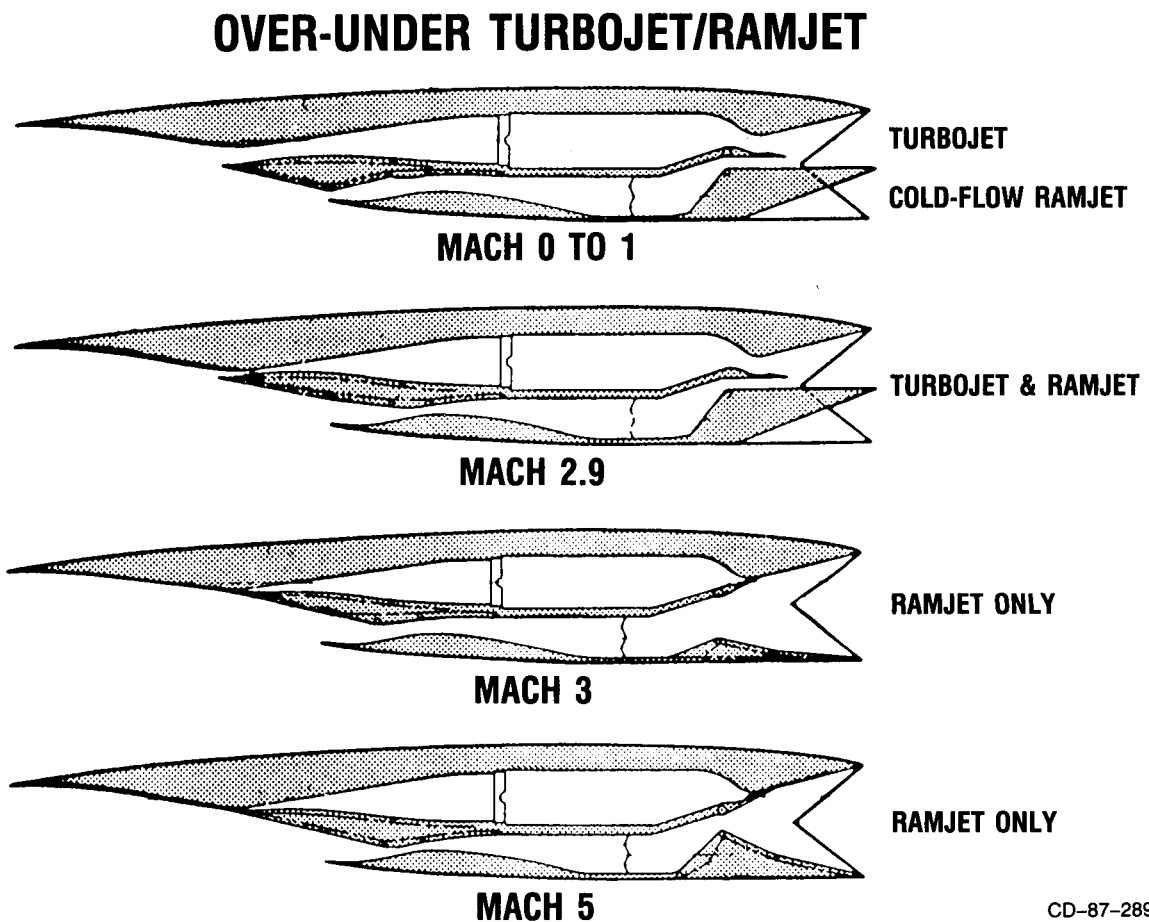
MACH 5 CRUISE AIRCRAFT STUDY



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OVER-UNDER TURBOJET/RAMJET

The various modes of operation for the over-under turbojet plus ramjet are illustrated in the figure. At subsonic flight speeds, the turbojet powers the aircraft, with cold flow through the ramjet. Near Mach 1.0 the ramjet is ignited, and both systems are operating until the aircraft approaches Mach 3.0. Between Mach 2.5 and 3.0, the turbojet spools down and the upper duct of the inlet is closed off. From Mach 3.0 to cruise speed, the aircraft is powered by the ramjet only.



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DESIGN METHODOLOGY

The ramjet (lower duct) portion of the inlet was designed by using a conventional method of characteristics (MOC) approach with boundary layer correction. One of the main driving factors in the design was length minimization, so as to reduce the weight as much as possible. The cowl shock is to be cancelled at the inlet shoulder, and the design throat Mach number is 1.6 inviscidly, which is reduced to approximately 1.2 when boundary layer corrections are made. The compression split is about 85 percent external (with four ramps) and 15 percent internal. The inlet employs variable ramp geometry for off-design operation.

DESIGN METHODOLOGY

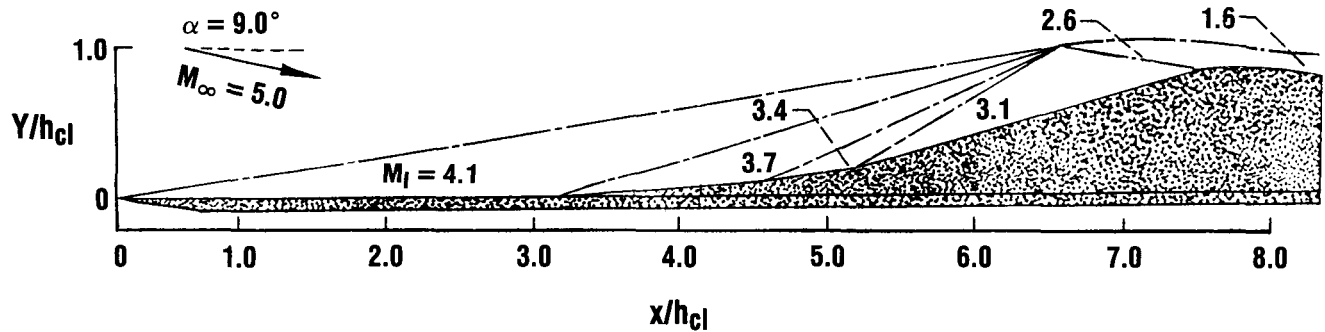
- DESIGNED BY METHOD OF CHARACTERISTICS WITH BOUNDARY LAYER CORRECTION.
- DESIGN MACH NUMBER = 5.0
- DESIGNED FOR MINIMUM LENGTH DUE TO WEIGHT CONSIDERATIONS.
- COWL OBLIQUE SHOCK CANCELLED AT RAMP SHOULDER.
- THROAT MACH NUMBER AT DESIGN CONDITIONS: INVISCID $M_T = 1.6$
WITH B.L. $M_T = 1.2$
- COMPRESSION: EXTERNAL (85 PERCENT): 4 RAMPS
INTERNAL (15 PERCENT): COWL SHOCK
DISTRIBUTED ISENTROPIC COMPRESSION
TERMINAL SHOCK
- VARIABLE RAMP GEOMETRY FOR OFF-DESIGN OPERATION

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Mach 5 INLET AERODYNAMIC DESIGN CONFIGURATION

The aerodynamic design of the inlet is shown. X- and Y- dimensions are nondimensionalized to cowl lip height. Mach numbers in the various flow regions are shown for the cruise (Mach 5) condition. At cruise conditions, the initial wedge angle of the inlet is at 9° .

MACH 5 INLET AERODYNAMIC DESIGN CONFIGURATION



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Mach 5 TWO-DIMENSIONAL INLET CHALLENGES

At high supersonic/low hypersonic speeds, several challenges face the inlet designer. The problems of multiple modes of operation and dual flow must be solved. High temperatures must be considered. Shock stability is not easily achieved; and the effects of three-dimensional flow, including corner flow and sidewall shock boundary layer interaction require complex bleed control systems. Inlet bleed and leakage cause serious performance penalties at the high speed conditions. Drag at off-design conditions is a problem, and weight reduction is a constant concern.

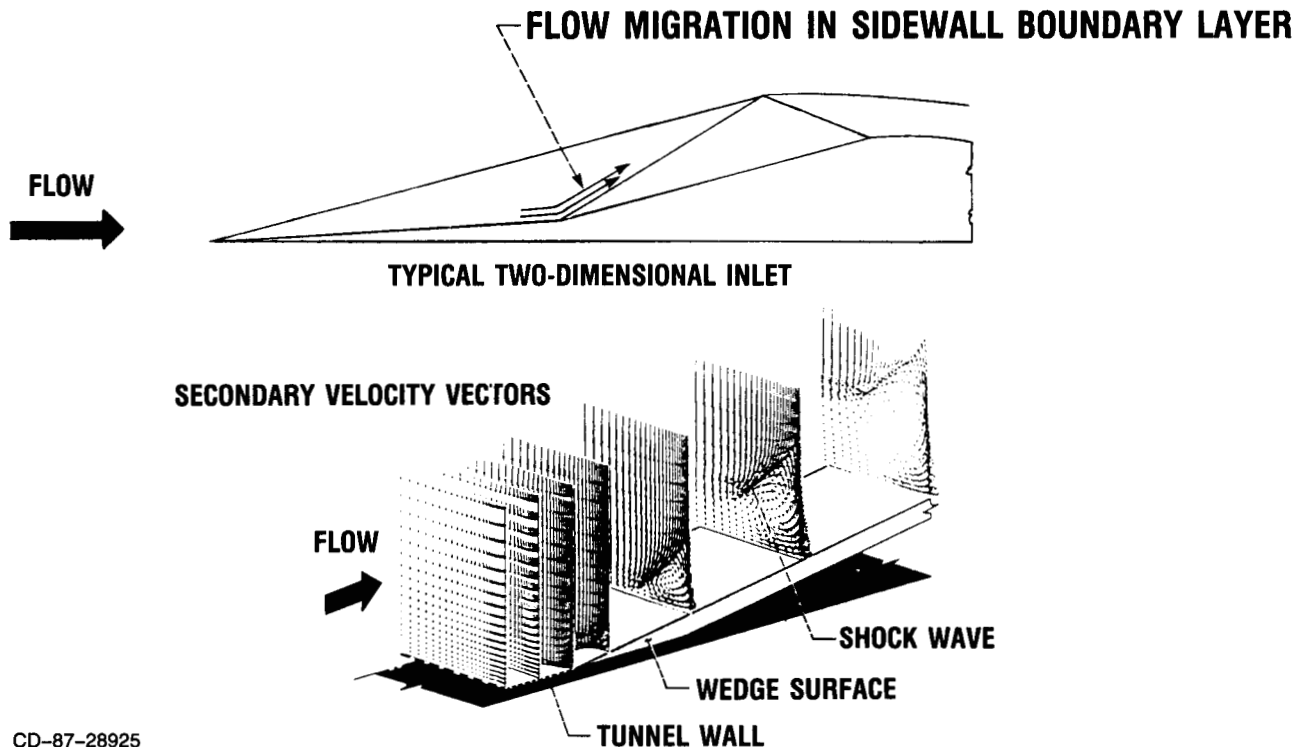
MACH 5 TWO-DIMENSIONAL INLET CHALLENGES

- **MULTIPLE MODE/DUAL FLOW OPERATION**
- **HIGH TEMPERATURE**
- **SHOCK SYSTEM STABILITY**
- **GLANCING SIDEWALL SHOCK/BOUNDARY LAYER INTERACTIONS**
- **THREE-DIMENSIONAL CORNER FLOWS**
- **INLET BLEED AND LEAKAGE PENALTIES**
- **WEIGHT REDUCTION**
- **OFF-DESIGN DRAG**
- **BOUNDARY LAYER CONTROL**

SIDEWALL BOUNDARY LAYER-GLANCING SHOCK WAVE INTERACTION

One problem with two-dimensional inlets is three-dimensional flow effects. Basic studies and analysis have shown that flow migration occurs in the boundary layer near regions where an oblique shock wave impinges on an adjacent sidewall. With thick boundary layers on an inlet sidewall, the higher pressure downstream of a sidewall-glancing shock can influence the upstream flow through the subsonic boundary layer. The result of this influence is a migration of the boundary layer along the oblique shock rather than parallel to the downstream ramp surface. A simple sketch showing the migration and analytical results are shown. This flow migration has a cumulative effect, with large regions of low-energy flow sweeping up the sidewalls as the flow moves aft toward the cowl.

SIDEWALL BOUNDARY LAYER-GLANCING SHOCK WAVE INTERACTION

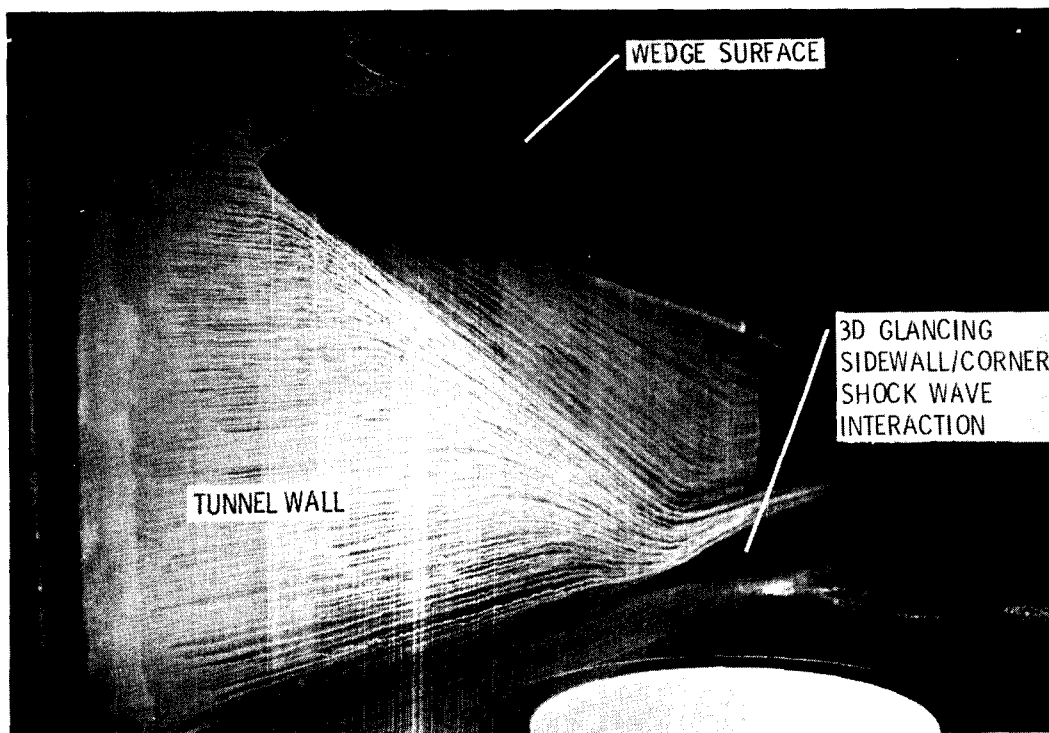


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THREE-DIMENSIONAL SIDEWALL SHOCK INTERACTION
EXPERIMENTAL SURFACE OIL FILM PATTERNS

The figure shows a simple compression wedge installed on the flat wall of a supersonic wind tunnel. The oblique shock from this wedge interacts with the tunnel boundary layer which has a thickness of approximately one inch. The surface oil film shows flow patterns in the boundary layer. The oil flows indicate that the boundary layer on the wall ahead of the oblique shock and a large portion of the boundary layer flow aft of the shock is influenced and is turned in a direction along the oblique shock angle rather than parallel to the wedge surface.

THREE-DIMENSIONAL SIDEWALL SHOCK INTERACTION
EXPERIMENTAL SURFACE OIL FILM PATTERNS

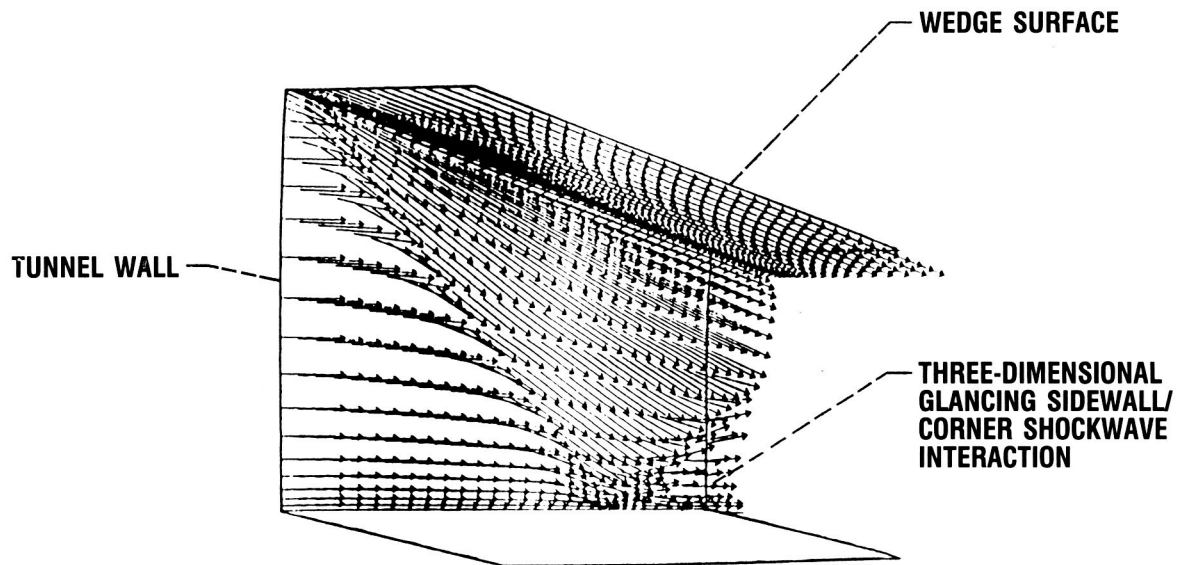


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THREE-DIMENSIONAL SIDEWALL SHOCK INTERACTION
ANALYTICAL SURFACE OIL FILM PATTERNS

A three-dimensional parabolized Navier-Stokes (PEPSIS code) solution for the simple case shown in the previous figure is presented below. The velocity vectors show that the boundary layer flow tends to turn in the direction of the shock wave.

THREE-DIMENSIONAL SIDEWALL SHOCK INTERACTION
ANALYTICAL SURFACE OIL FILM PATTERNS

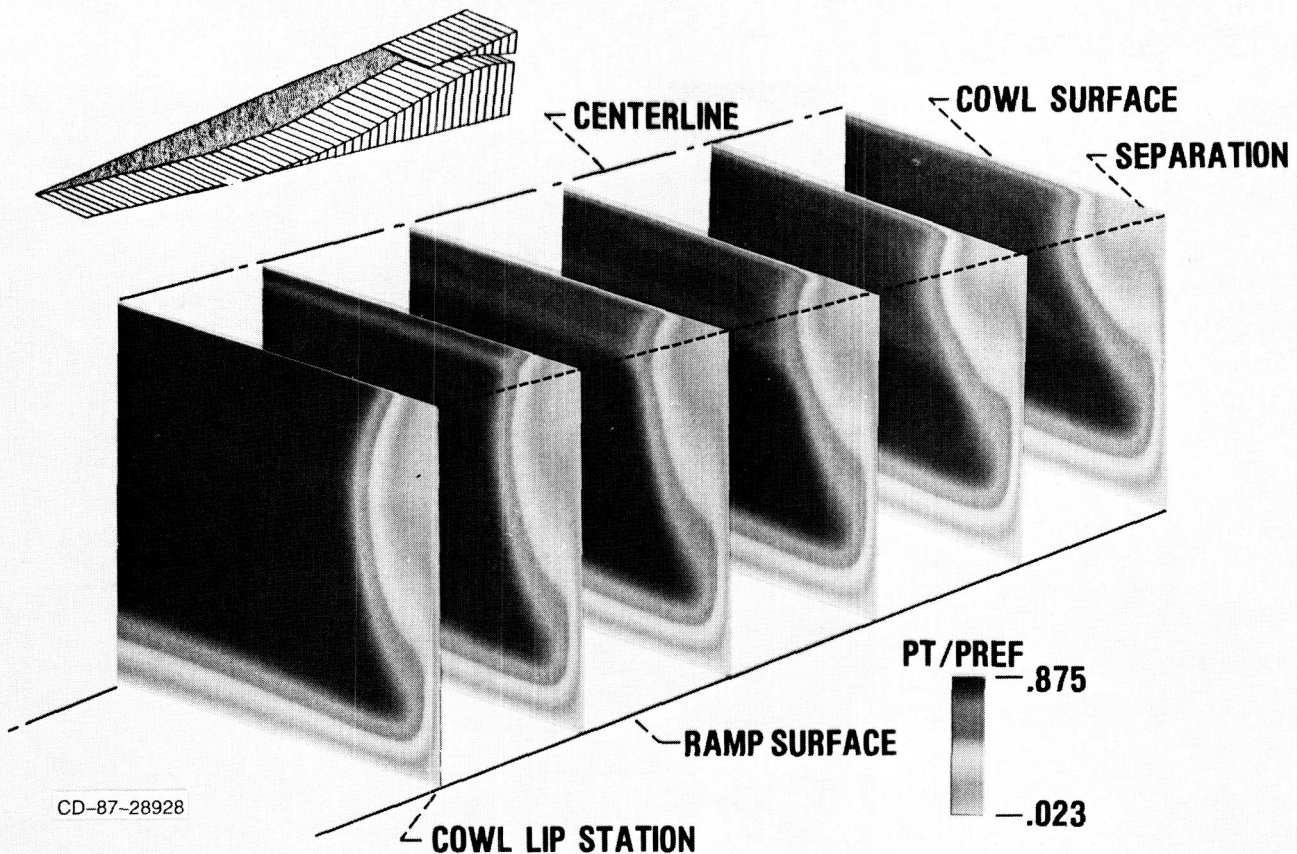


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LEWIS HIGH-SPEED INLET PROGRAM
TOTAL PRESSURE DISTRIBUTION

PEPSIS analyses have shown that sidewall shock boundary layer interactions will have a dominant effect on the Mach 5 inlet. The figure shows total pressure distributions on cross-planes at several stations in the inlet aft of the cowl lip. Only half-planes are shown, as flow is symmetrical. Low energy flow swept up the sidewalls is captured by the cowl, resulting in a large separation region in the cowl-sidewall corner at a location forward of the inlet ramp shoulder. Such large separations will cause the inlet to unstart (expel the terminal shock out of the inlet), and thus are a major concern.

LEWIS HIGH-SPEED INLET PROGRAM
TOTAL PRESSURE DISTRIBUTION

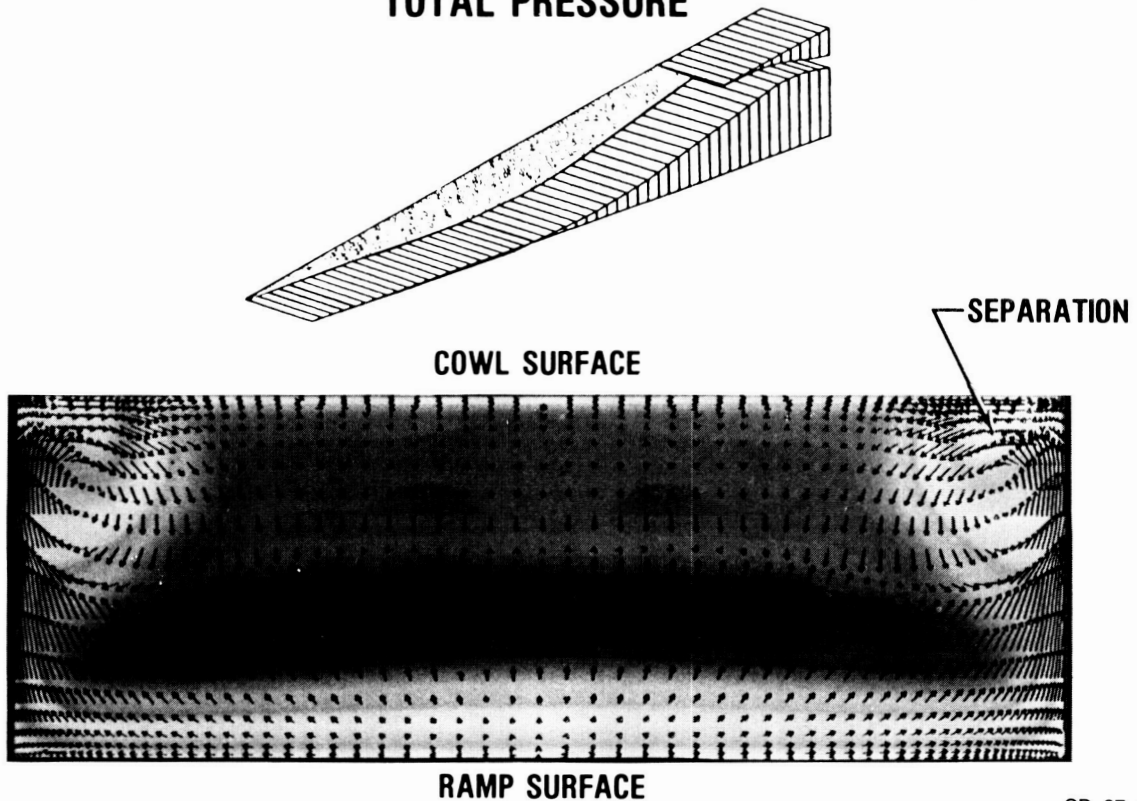


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LEWIS HIGH-SPEED INLET PROGRAM
TOTAL PRESSURE

The figure shows the total pressure distribution for a cross-section of the inlet just aft of the cowl lip. Velocity vectors are superimposed on this pressure distribution, showing recirculation in the corner separation region. The influence of the cowl oblique shock on the mid-stream flow can also be seen.

LEWIS HIGH-SPEED INLET PROGRAM
TOTAL PRESSURE

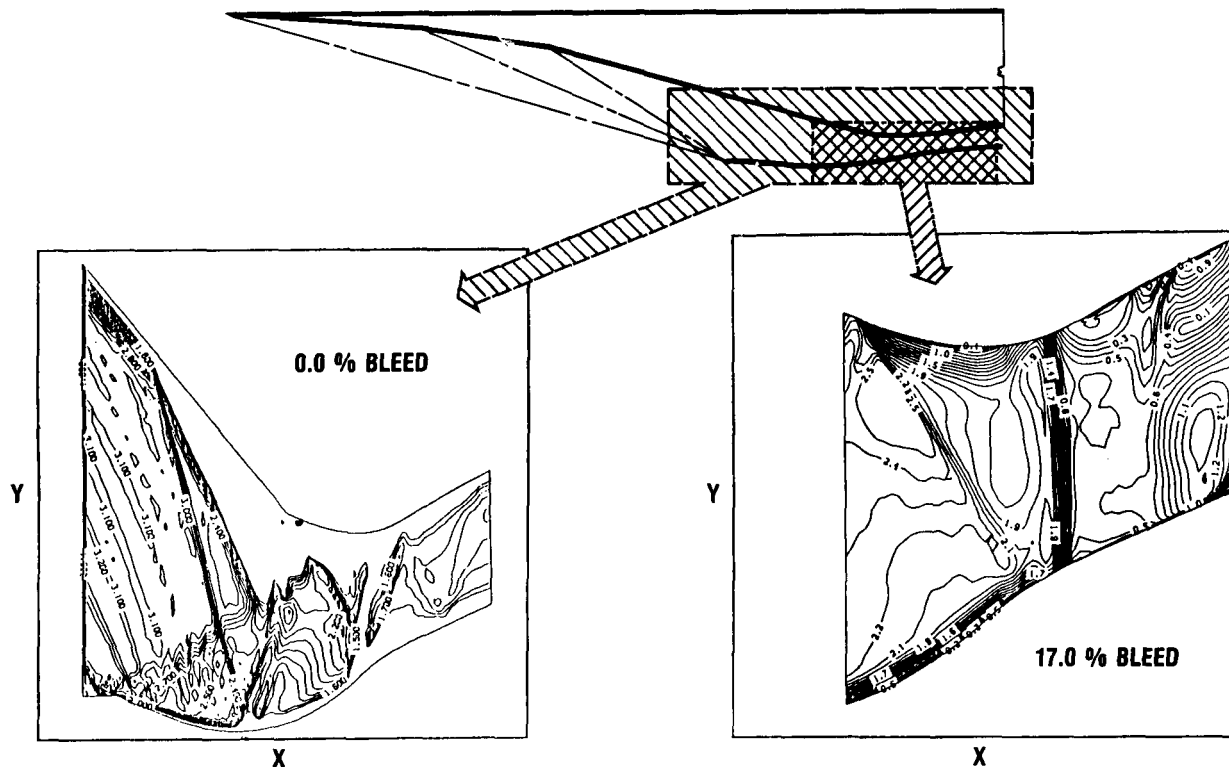


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TWO-DIMENSIONAL KUMAR CODE ANALYSIS: Mach NUMBER CONTOURS

A two-dimensional analysis of the Mach 5 inlet, showing the benefits of boundary layer bleed on the ramp and cowl was carried out by W. Rose and E. Perkins of Rose Engineering and Research, consultants to Lockheed on the Mach 5 project. The analytical code was developed by A. Kumar of the NASA Langley Research Center. The Mach number distribution for the no-bleed case shown on the left represents the area encompassed by the large dashed box in the inlet sketch at the top. These contours show very large, separated regions on the ramp. The inlet will not operate in a started mode with these conditions, but would operate with the normal shock external to the cowl lip. The Mach contours on the right, with 17 percent bleed distributed through the inlet, represent the area in the inlet throat region on the sketch above. Bleed has prevented the large separations, and a normal shock has been stabilized in the inlet.

TWO-DIMENSIONAL KUMAR CODE ANALYSIS: MACH NUMBER CONTOURS



CD-87-28930

BENEFITS TO BE OBTAINED FROM INLET TEST PROGRAM

A 1/3 scale model of the Mach 5 inlet is to be tested in the NASA Lewis 10 x 10 SWT. The main goals of this test plan are to: (1) provide data for code validation and for the development of inlet design codes, and (2) determine overall inlet performance and bleed requirements.

BENEFITS TO BE OBTAINED FROM INLET TEST PROGRAM

PROVIDE FUNDAMENTAL AND DESIGN DATA FOR CODE VALIDATION

- **GLANCING SIDEWALL SHOCK/
BOUNDARY LAYER INTERACTION**
- **THICK BOUNDARY LAYER/MULTIPLE
AND OBLIQUE SHOCK INTERACTION**
- **THICK BOUNDARY LAYER/NORMAL
SHOCK INTERACTION**
- **CORNER FLOW**
- **INLET DESIGN CODE DEVELOPMENT
DATA**

DETERMINE OVERALL INLET PERFORMANCE

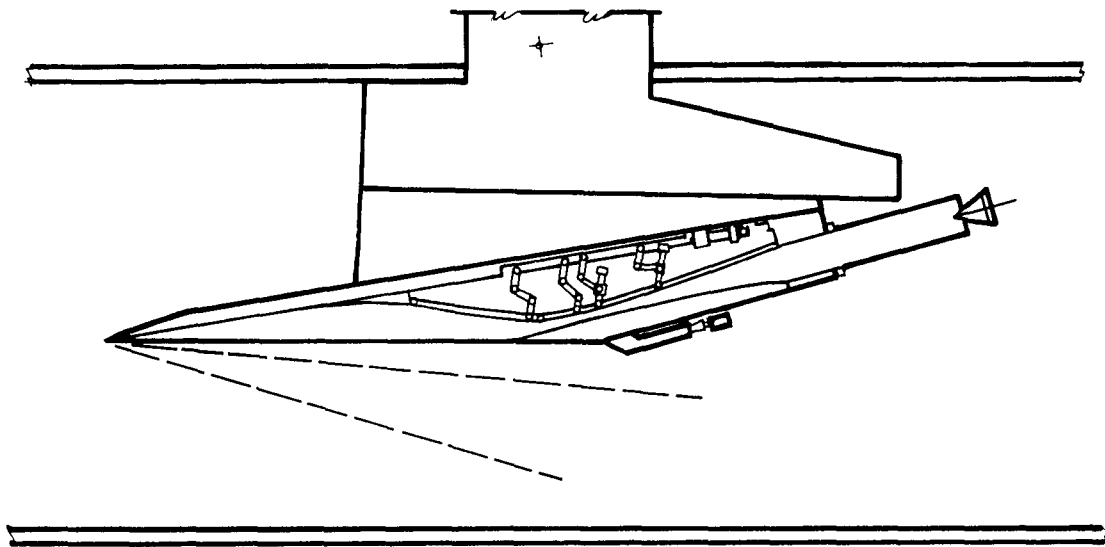
- **DESIGN AND OFF-DESIGN
AERODYNAMIC CHARACTERISTICS**
- **BLEED REQUIREMENTS**
- **UNSTART/RESTART CHARACTERISTICS**
- **CONTROL DATA SIGNALS**
- **VERIFICATION OF INLET DESIGN
TECHNIQUE**
- **VERIFICATION OF ACCELERATOR
PLATE TEST TECHNIQUE**

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INLET RESEARCH

The Lewis 10 x 10 SWT has a maximum Mach number capability of 3.5. Therefore, to test the inlet at design conditions, the Mach 5.0 design freestream flow which is compressed to Mach 4.1 on the first ramp, is to be simulated by expanding the Mach 3.5 tunnel flow to Mach 4.1 underneath a plate at negative angle of attack. This "accelerator plate" test technique duplicates the actual inlet flow conditions with the exception of the initial oblique compression shock. The data will be corrected for the total pressure loss for this initial compression.

INLET RESEARCH



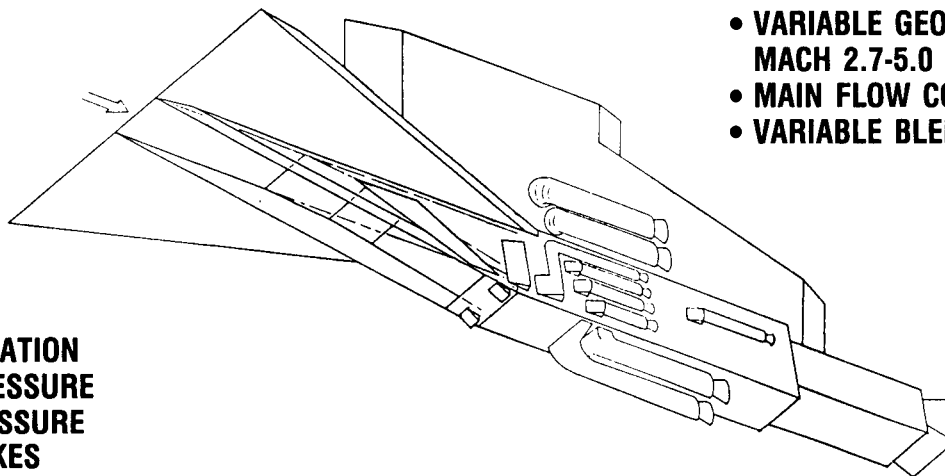
TWO-DIMENSIONAL INLET INSTALLED IN 10 x 10 SWT

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MACH 5 INLET MODEL

An isometric sketch of the model to be tested in the NASA Lewis 10 x 10 SWT is shown. The model incorporates remotely variable ramp geometry, main duct mass-flow control, and bleed exit plugs. The model is extensively instrumented with static pressure taps, total pressure rakes, translating flow angularity probes, and dynamic pressure transducers. It is a very large model, with the accelerator plate being 100 in. wide and an overall model length of about 20 ft. The cowl lip height is 16 in. with a capture width of 16 in.

MACH 5 INLET MODEL



REMOTE CONTROL:

- VARIABLE GEOMETRY RAMP
MACH 2.7-5.0
- MAIN FLOW CONTROL
- VARIABLE BLEED EXITS (15)

INSTRUMENTATION

- STATIC PRESSURE
- TOTAL PRESSURE
 - FIXED RAKES
 - TRANSLATING PROBES
- DYNAMIC PRESSURE TRANSDUCERS
 - STATIC
 - TOTAL

DIMENSIONS

- PLATE WIDTH = 100 in.
- COWL LIP HEIGHT = 16 in.
- LENGTH = 20 ft.
- CAPTURE WIDTH = 16 in.

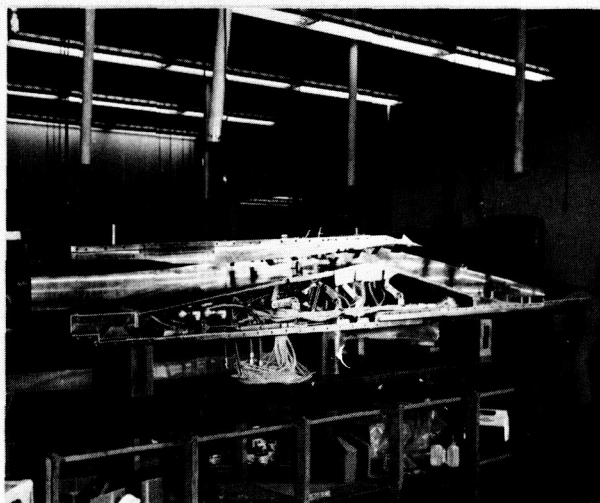
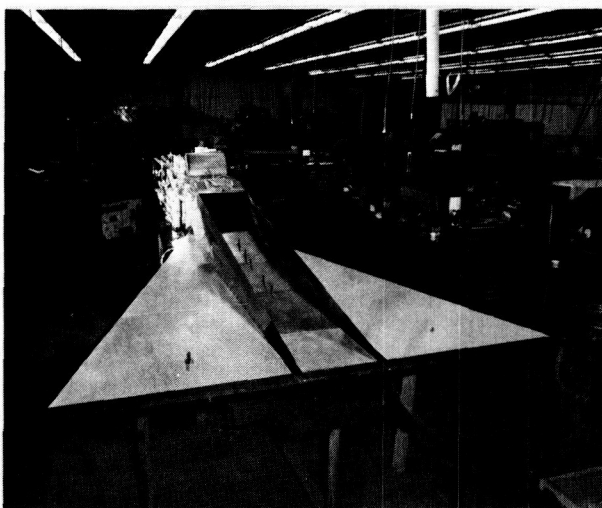
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MACH 5 INLET TEST MODEL

Two views of the Mach 5 inlet model are shown. The model is made of stainless steel, except for the accelerator plate, which is aluminum. On the right, a side view of the inlet is shown with the sidewall removed to show variable ramp mechanisms. A single, large pair of actuators raises and lowers all moveable sections of the ramp simultaneously. The inlet duct is entirely two-dimensional, from leading edge to mass-flow control plug. All bleed regions are compartmentalized to prevent recirculation. Collapsible bellows are used to duct the compartmentalized ramp bleed through the ramp plenum.

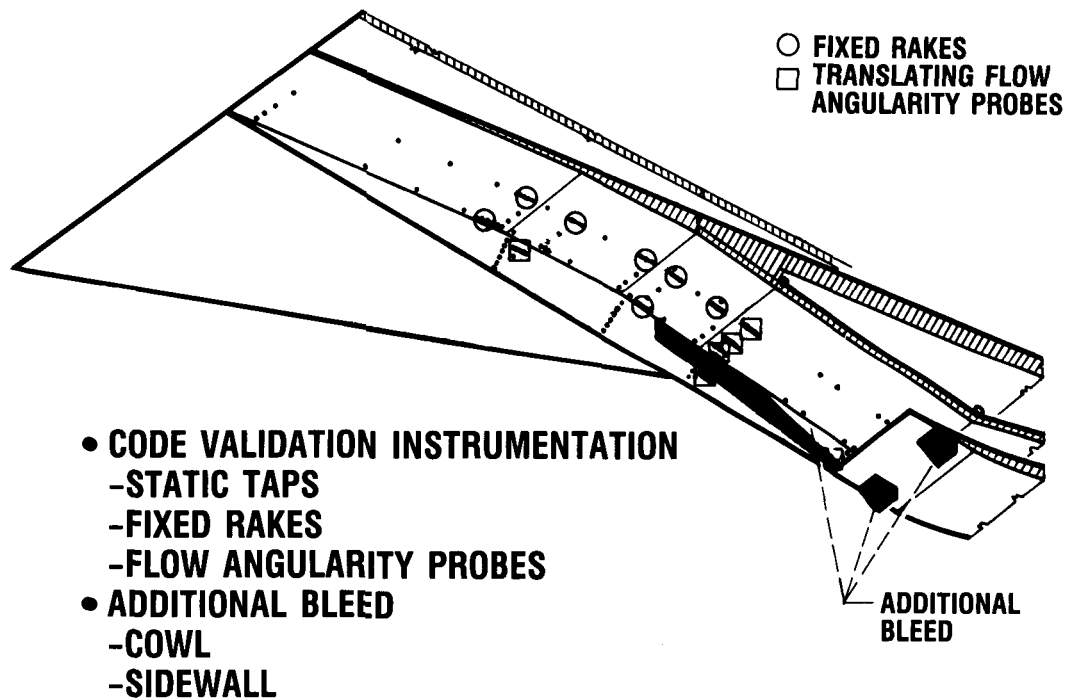
MACH 5 INLET TEST MODEL



IMPACT OF "PEPSIS" ANALYSIS ON MODEL INSTRUMENTATION AND BLEED

As a result of the computational analysis indicating separations due to boundary layer migration from sidewall shock boundary layer interactions, modifications were made to the original model design. Additional instrumentation was added on the ramp and sidewall to map the flow migration phenomena and, also, to provide some code validation data. Additional bleed regions were added on the inlet sidewalls and on the cowl in the corners just aft of the cowl lip. This bleed was added to help control and/or eliminate corner separations.

IMPACT OF "PEPSIS" ANALYSIS ON MODEL INSTRUMENTATION AND BLEED

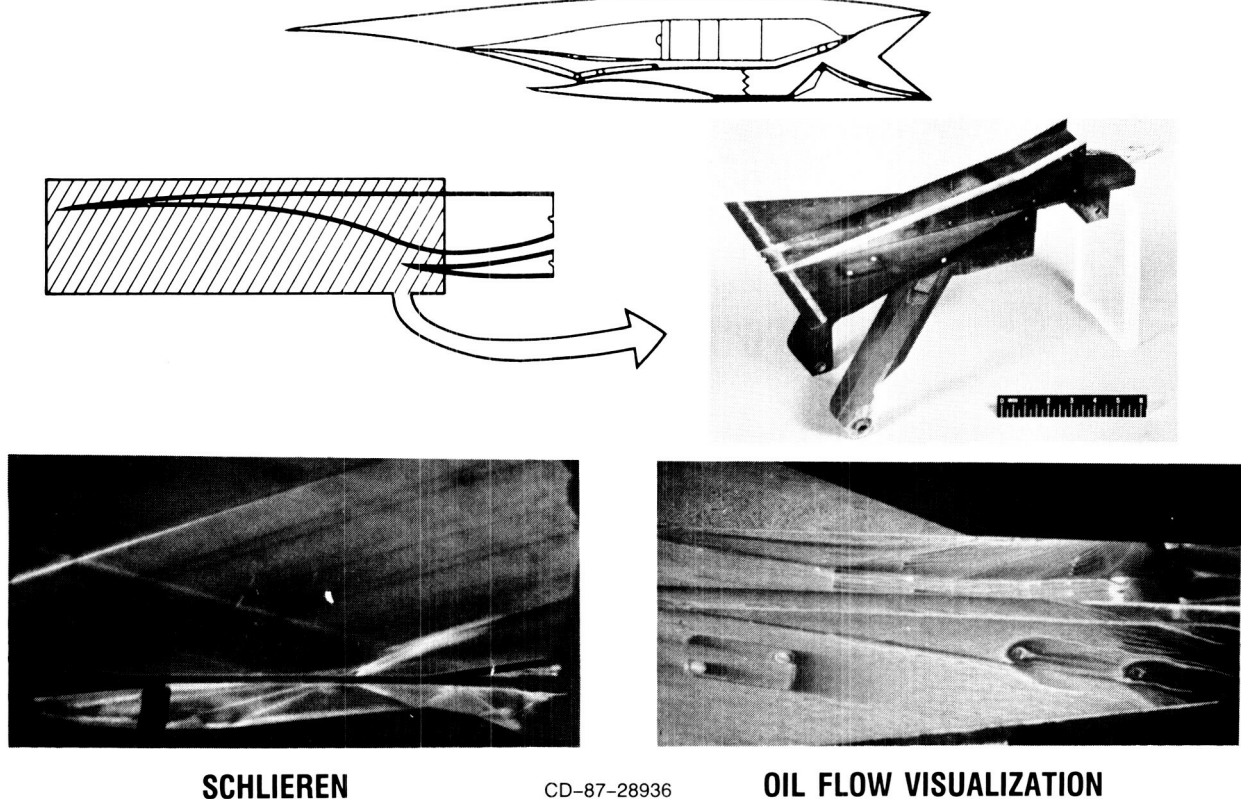


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SMALL-SCALE PLATE/RAMP MODEL: 1 X 1 SWT

A simple, small-scale model of the high-speed inlet was tested in the NASA Lewis 1 x 1 SWT to provide some calibration data for comparison with analytical predictions. This model had capture dimensions of 1.6 in. by 1.6 in. and duplicated the inlet geometry to the cowl lip and ramp shoulder. Aft of these stations, the inlet was opened-up to allow inlet starting. The Schlieren photo on the lower left is for design flow conditions, and oblique shock waves from the second, third, and fourth ramps, as well as the cowl shock can be seen. The surface oil film photograph at the right is for an off-design Mach number (Mach 3.0 on the first ramp). For the condition shown in the photograph, the inlet was unstarted, as indicated by the ramp flow near the cowl lip station. However, upstream of this location the sidewall boundary layer flow migration that results from the boundary layer-glancing shock interaction can be seen.

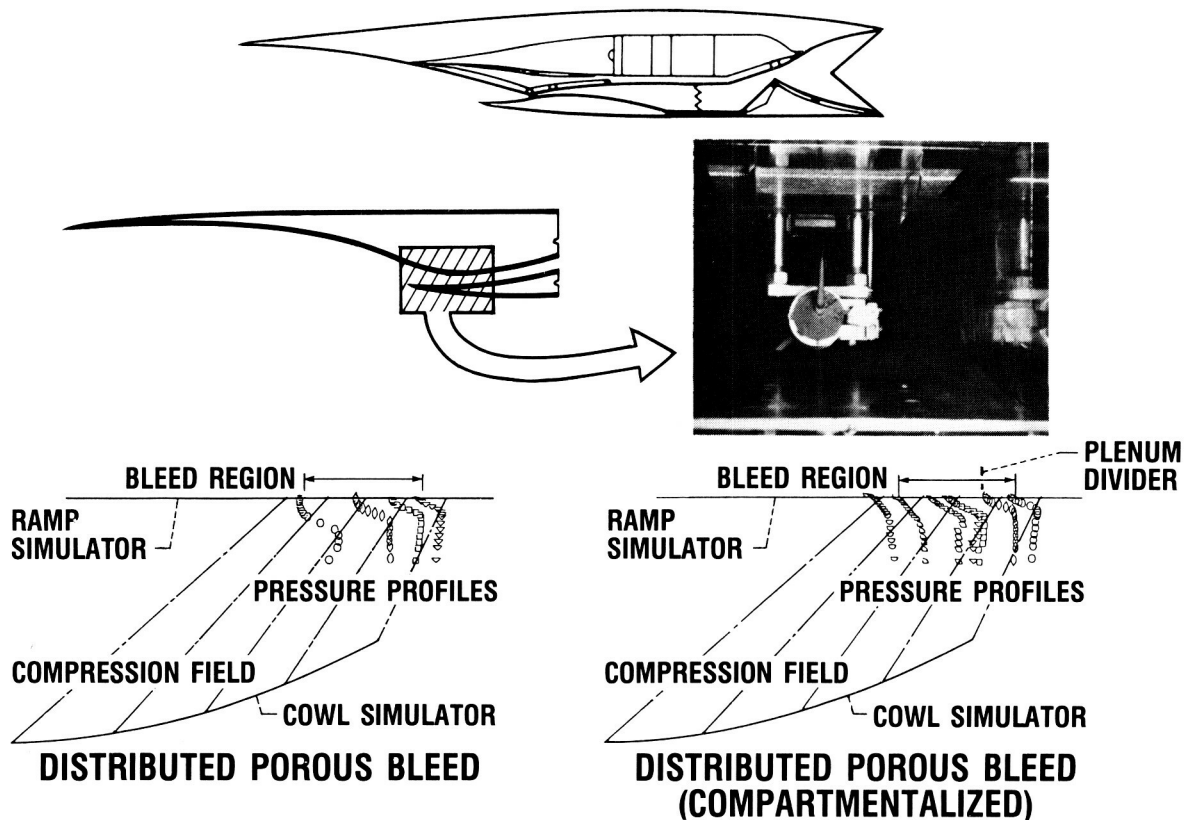
SMALL-SCALE PLATE/RAMP MODEL: 1 X 1 SWT
TURBOJET AND RAMJET



BLEED CONTROL STUDIES: 1 X 1 SWT

An aerodynamic design approach that may be used to reduce the weight of an inlet is to decrease the length over which the distributed cowl compression intersects the ramp surface. This is accomplished by increased curvature of the cowl and results in a large pressure rise over a short distance on the ramp. However, these large pressure gradients with large approach boundary layers can result in separation. A simple experimental program was conducted to study ramp bleed configurations to control the interaction of the pressure gradient-boundary layer in this region. For this test, the cowl was simulated by a contoured compression plate and the ramp by the tunnel wall (photo on the upper right in the figure). The tunnel wall incorporated a bleed plate in which various bleed patterns could be studied. A translating probe was used to survey the flow field. Surveys for a distributed porous bleed configuration are shown. Note that a separation occurs in the left-hand figure, due to recirculation in the bleed plenum. When the bleed is compartmentalized (as shown on the right) boundary layer is successfully controlled.

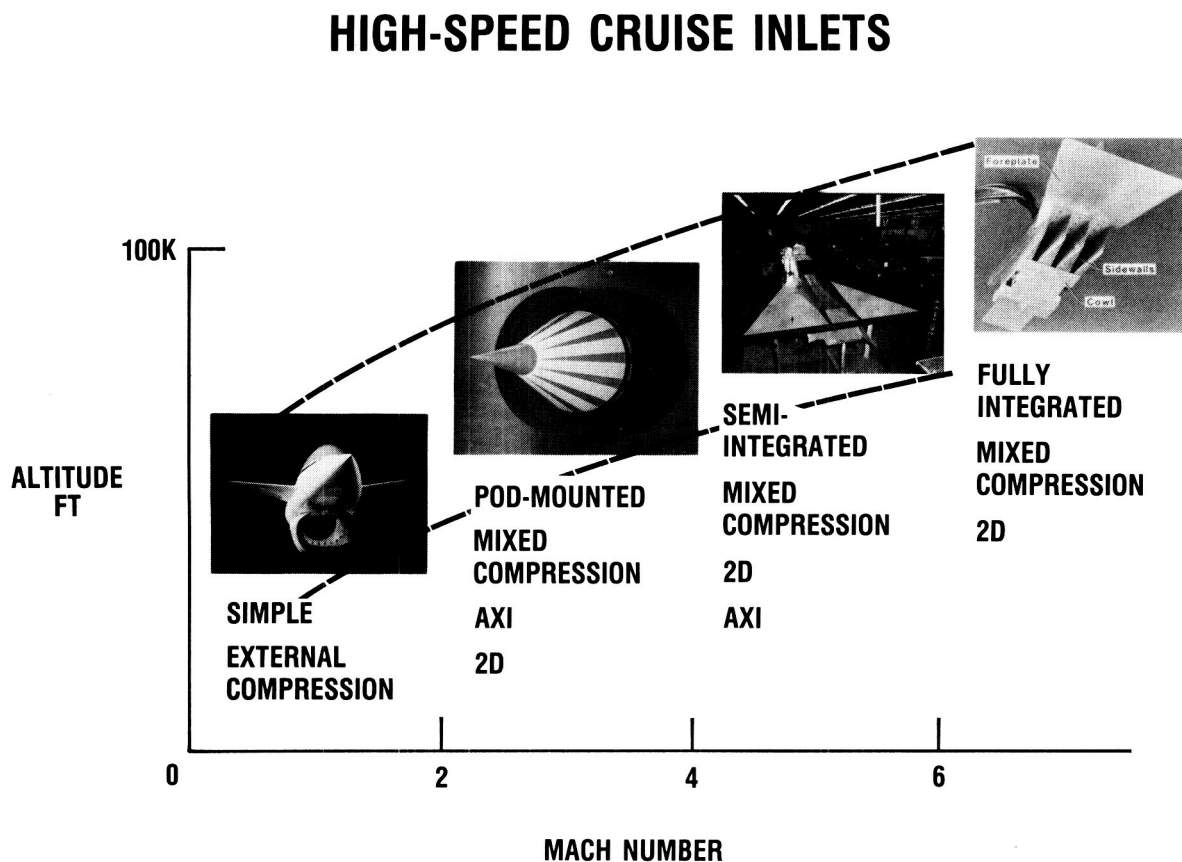
BLEED CONTROL STUDIES: 1 X 1 SWT



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HIGH-SPEED CRUISE INLETS

The figure shows a plot of flight Mach number compared to altitude. Wind tunnel models of four representative inlets for different Mach numbers and altitudes are shown in the photographs. In the low supersonic speed range, inlets tend to be simple in geometry, and often have entirely external compression. The external compression HiMAT inlet is shown on the left. For aircraft operating in the Mach 2 to 4 range, inlets are usually pod-mounted and have mixed compression. Such an inlet, with axisymmetric geometry, is shown in the second photo from the left. This inlet incorporates a collapsing variable-diameter centerbody. In the Mach 4 to 6 range, inlets tend to be more integrated into the airframe. A two-dimensional Mach 5 inlet model is shown in the third photo from the left. Inlets for hypersonic (Mach 6+) aircraft are fully integrated into the airframe, and are normally two-dimensional rather than axisymmetric. A scram inlet model with sidewall compression is shown in the photograph at the right.



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HYPersonic PROPULSION RESEARCH

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Langley Research Center

ABSTRACT

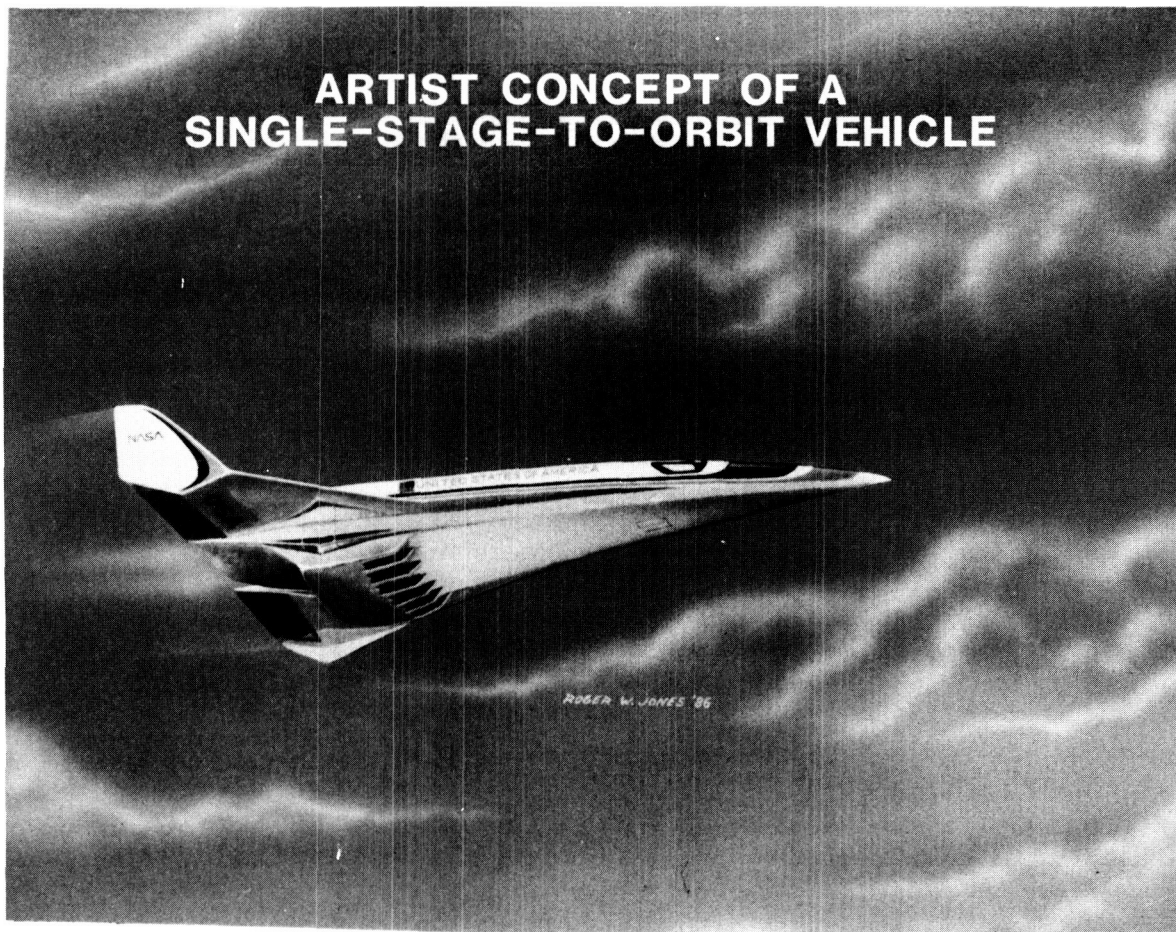
The NASA Langley Research Center has been conducting hypersonic propulsion research since the 1960's. In 1965, the Hypersonic Research Engine (HRE) project was undertaken to demonstrate internal performance of a scramjet with the X-15 as a test platform. The X-15 program was terminated in 1968 before the scramjet was ready for flight testing. However, two regeneratively cooled models were constructed and the internal performance of the engine was demonstrated in ground tests at the Lewis Research Center's Plum Brook test facility. Following the HRE program, the Hypersonic Propulsion Branch developed a modular airframe-integrated scramjet concept. This engine concept yields higher installed performance by reducing drag associated with mounting the engine. The airframe-integrated scramjet uses the vehicle forebody as a part of the compression surface, attaches engine modules directly to the underside of the vehicle, and uses the aft end of the airframe as a nozzle expansion surface. The development of technology for the modular airframe-integrated scramjet has been the focus of this research for the past several years.

As part of this research, a variety of inlet concepts have been explored and characterized. The emphasis of the inlet program has been the development of the short (light weight), fixed geometry, side-wall-compression inlets that operate efficiently over a wide Mach number range. As hypersonic combustion tunnels were developed, programs to study the parameters controlling fuel mixing and combustion with single and multiple strut models were conducted using direct connect test techniques. These various tests supported the design of subscale engine test hardware that integrated inlet and combustor technology and allowed the study of the effect of heat release on thrust and combustor/inlet interaction. A number of subscale (8-in.-high by 6-in.-wide) engine tests have demonstrated predicted performance levels at Mach 4 and Mach 7 simulated flight conditions.

The attached material summarizes a few of the highlights from this research program.

ARTIST CONCEPT OF A SINGLE-STAGE-TO-ORBIT VEHICLE

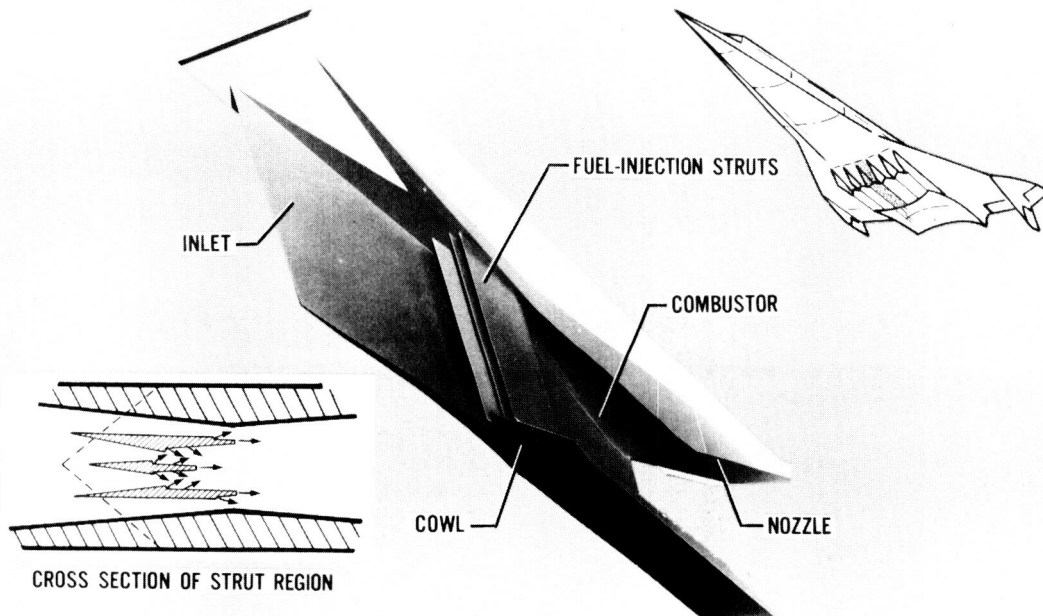
An artist concept of the current focus of much of the nations hypersonic research is shown. Since February 1985 when President Reagan announced that the country would build a National Aero-Space Plane (NASP) there has been an accelerated program at government and industrial laboratories to access and define the technology advances required to make single stage to orbit possible. NASA has teamed with the Defense Department, industry, and the universities to develop the technologies necessary to design and construct an experimental aircraft needed to demonstrate hypersonic flight by using airbreathing propulsion. The proposed research airplane, designated the X-30, is not an operational vehicle but a test bed to demonstrate the various technologies required for the design of three classes of vehicles--hypersonic transports that would cruise at Mach 5 or higher, space transports, and transatmospheric military vehicles.



AIRFRAME-INTEGRATED SUPERSONIC COMBUSTION RAMJET

This figure shows the Langley concept for an airframe-integrated supersonic combustion ramjet (scramjet). The engine modules are mounted on the underside of the vehicle and use the vehicle forebody as part of the nozzle compression. The compression is completed by the inlet sidewalls and the fuel injection struts. The use of fuel injection struts to complete the compression shortens the inlet and provides locations for distributing the fuel in the incoming compressed air. The relative amount of parallel and transverse injected fuel is used to control the mixing and thus the heat release distribution required for the desired performance as a function of flight Mach number. The modular design allows testing of a single engine in a smaller facility than that required if the total propulsion system had to be tested.

AIRFRAME-INTEGRATED SUPERSONIC COMBUSTION RAMJET

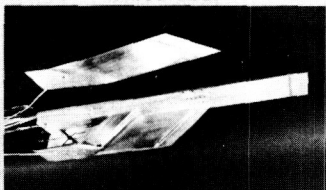


HYPERSONIC INLET DESIGN

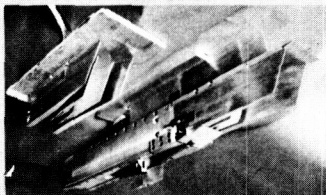
Several inlet designs have been constructed and evaluated at Mach 3 to 6. Currently the testing range is being extended to Mach 18 to support the NASP program. The three strut inlet with a swept combustor was the first inlet of this class completely characterized--mass capture, pressure recovery, contraction ratio, and throat Mach number as a function of free stream Mach number. Computational fluid dynamics is becoming more mature and is being used to design and evaluate alternate inlet designs like the rectangular to circular inlet and the two strut inlets shown. The three-module inlet model is being used to determine the effect of an inlet unstart on the performance of the remaining started inlets.

HYPERSONIC INLET DESIGN

INLET MODELS
Three-strut



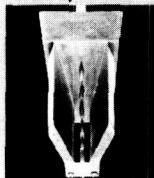
Two-strut



No strut



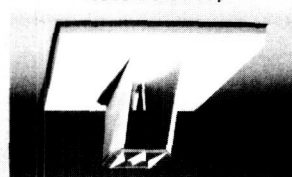
Step strut



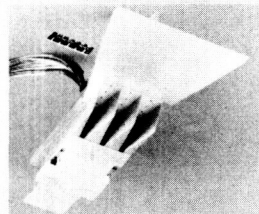
INLET RESEARCH

- Existing
 - Experimental data base for swept rectangular sidewall compression inlets
- Planned research
 - Performance $M = 3.5$ to 18
 - Inlet starting
 - Combustor/inlet interaction
 - Code validation $M = 3.5$ to 18 in air and helium
 - Highly-cooled-wall effects on inlet performance

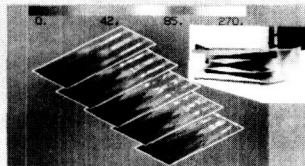
INLET MODELS
Reverse sweep



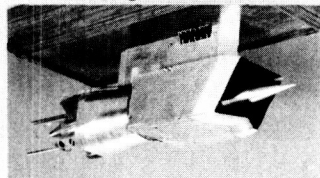
Three-module



CFD FLOWFIELD
Two-strut inlet



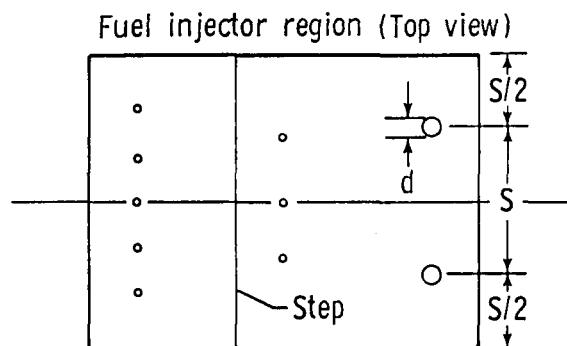
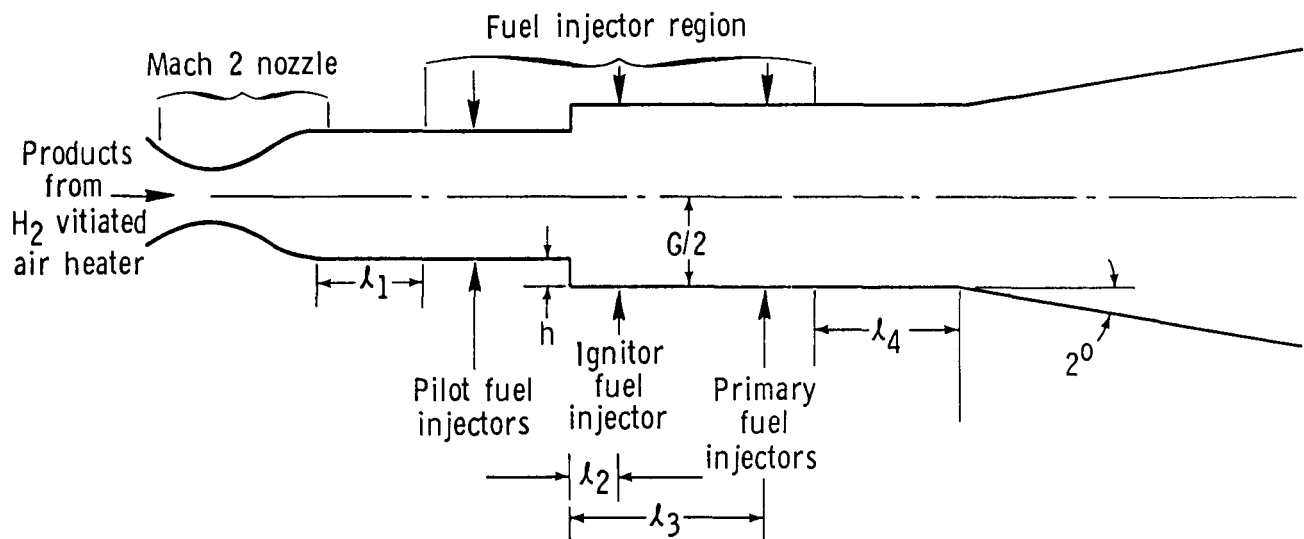
Rectangular to circular



SCHEMATIC OF DIRECT CONNECT TEST APPARATUS

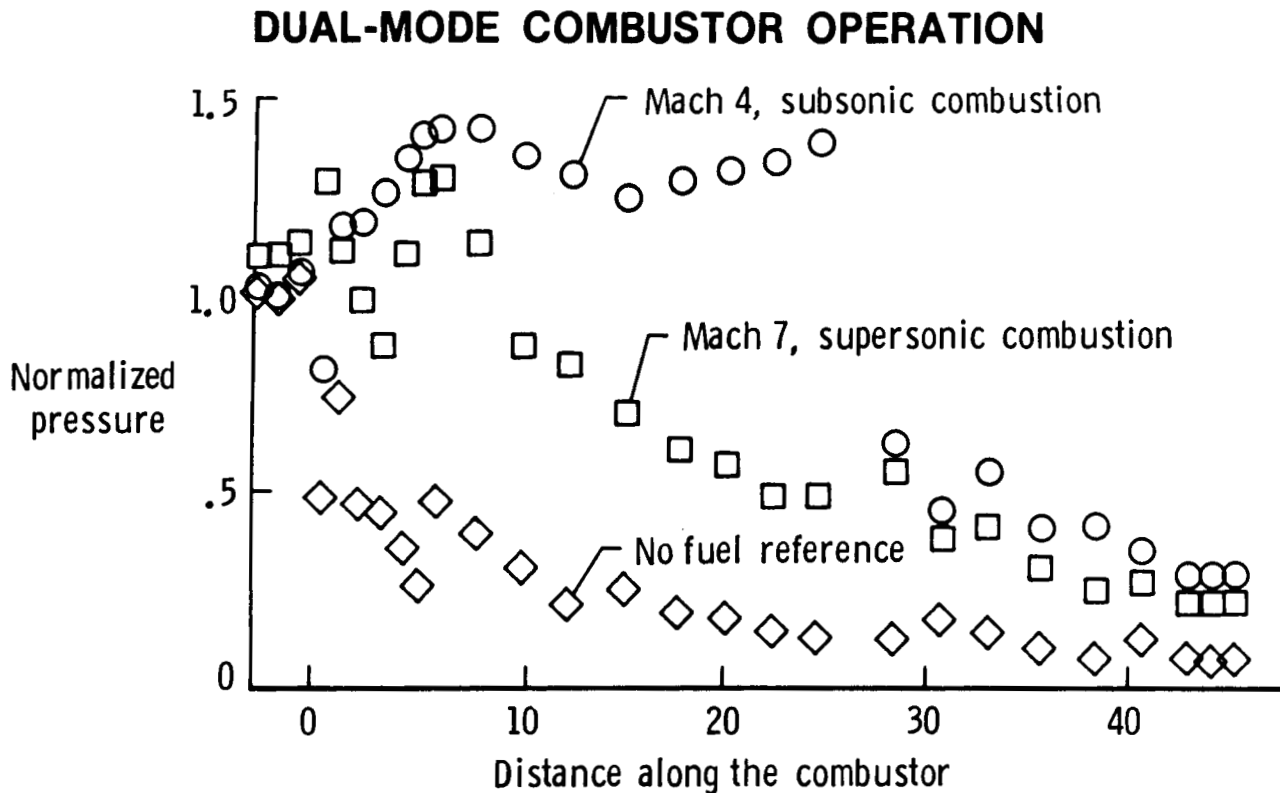
In order to reduce the size of the facility required to test a scramjet engine, direct connect tests of the combustor are conducted by simulating the inlet flow into the combustor region. Typical direct connect test hardware are shown below. This hardware is being used to determine combustor geometry effects on scramjet performance. The figure shows some of the geometry variables being investigated.

SCHEMATIC OF DIRECT CONNECT TEST APPARATUS



DUAL-MODE COMBUSTOR OPERATION

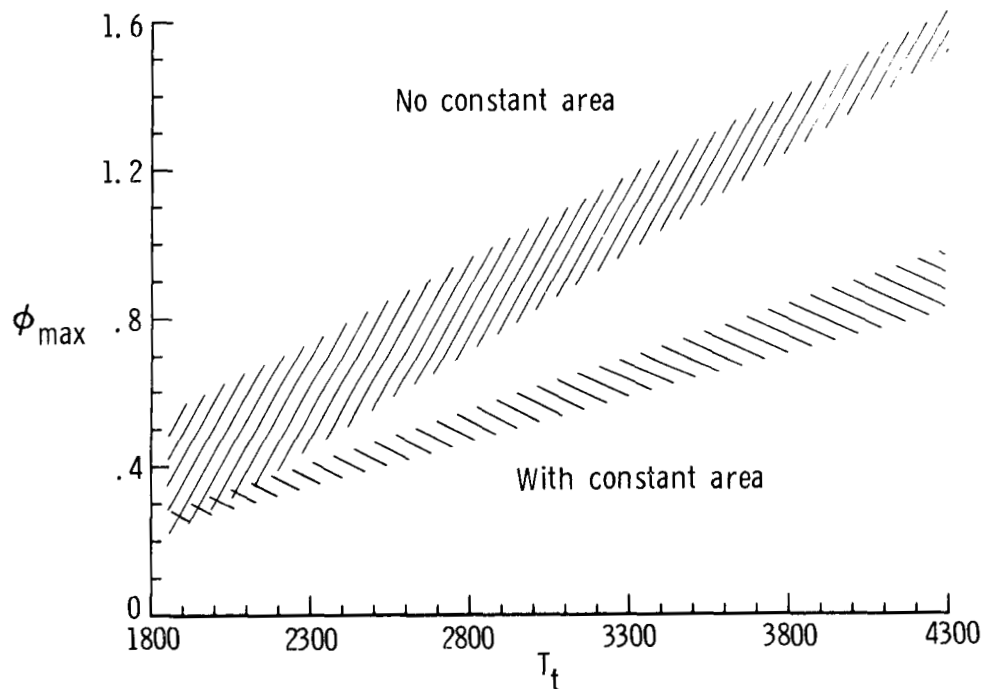
Using direct connect test hardware, the two different pressure distributions shown were obtained by varying the total temperature and the fuel injection location with the Mach number at the combustor entrance held constant at 1.7. The Mach 4 distribution with a major portion of the duct operating subsonic had most of the fuel injected parallel to the flow. For the Mach 7 total temperature data, most of the fuel was injected perpendicular to the air flow and much of the combustion occurred at supersonic conditions.



UPSTREAM INTERACTION LIMIT

This figure shows the maximum equivalence ratio that can be injected in the direct connect hardware as a function of total temperature. The Mach number entering the combustor from the facility nozzle was 2.0. Adding a constant area section 2.2 duct heights downstream of the fuel injector region reduced the amount of fuel that could be added without causing a pressure rise at the combustor entrance station. The duct divergence downstream of the fuel injectors of the constant area section was 2° on the top and bottom walls.

UPSTREAM INTERACTION LIMIT EFFECT OF CONSTANT AREA COMBUSTION

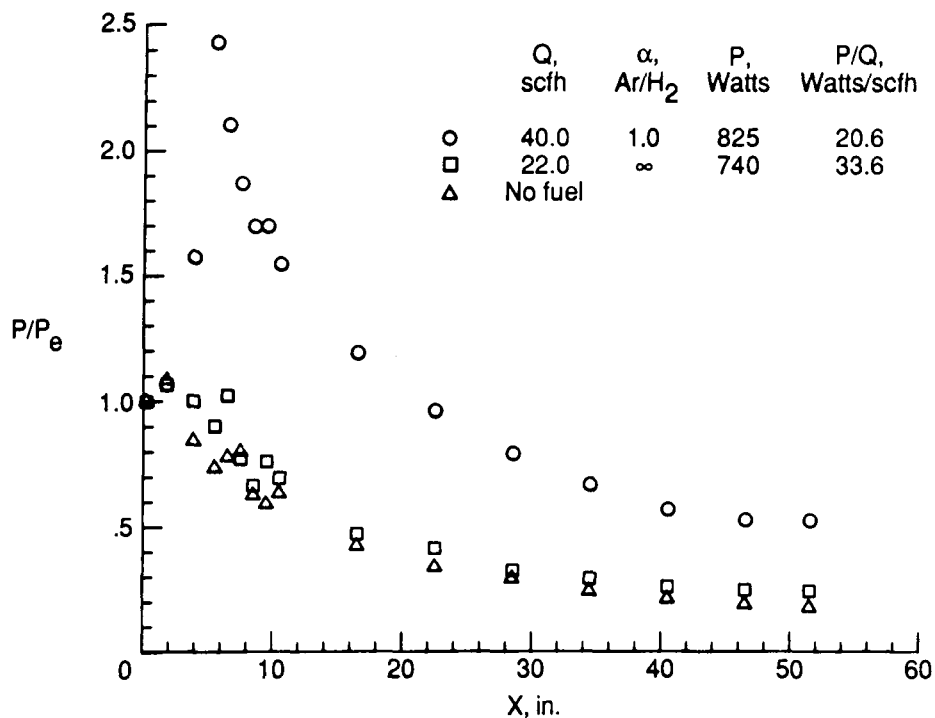


PRESSURE DISTRIBUTIONS FOR PILOTED TESTS

A number of test have been conducted to explore ignition and flameholding techniques for hypersonic combustion. At low flight speeds, the total temperature is not high enough for autoignition; at high speeds the reduced residence time may make flameholding difficult. One ignition technique explored was the use of an electrically driven plasma jet operating on a hydrogen/argon mixture. The figure shows the pressure distributions for a Mach 2 combustor operating below the autoignition temperature (total temperature 1400 °R). When 50 percent hydrogen/argon mixture was used in the plasma jet operating at less than 1 kw, the pressure distribution indicated considerable pressure rise. However, when the torch was operated on all argon the pressure distribution was similar to the no fuel distribution. These results indicate that the hydrogen atoms generated in the plasma, not the thermal heating of the gases were the ignition source.

PRESSURE DISTRIBUTIONS FOR PILOTED TESTS

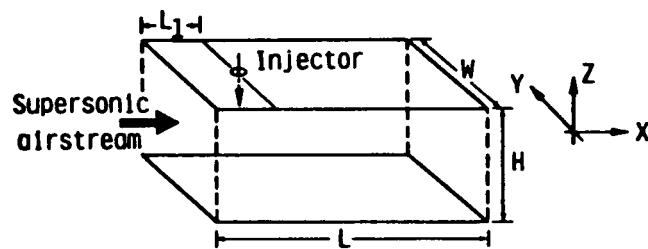
AR/H₂ AND ARGON PLASMAS, T_t = 1400 R, $\phi = 0.28$



COMPARISON OF CFD AND EXPERIMENTAL MIXING RESULTS

The use of large computers for the calculations of complex flows is a new tool available to the scramjet designer. In order for these codes to be used with confidence, validation tests with well defined boundary conditions must be conducted and compared with the corresponding computed results. This figure shows that the result of one such experiment where a sonic transverse jet was injected into supersonic Mach 2.06 flow. The lines are the calculated mass contours and the dots are the edge of the jet as defined by laser induced fluorescence of iodine. This favorable comparison indicates that the CFD code could be extended to other similar geometries with confidence. In fact, the code has been modified to include chemical reactions and similar reacting flow experiments are planned.

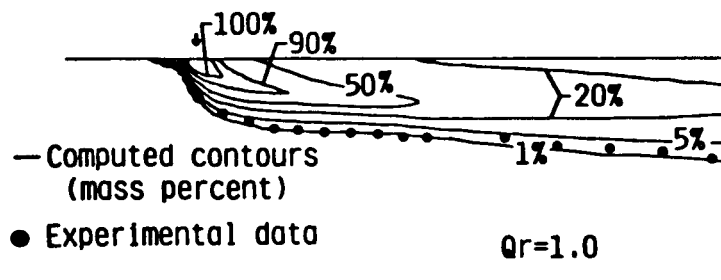
COMPARISON OF CFD AND EXPERIMENTAL MIXING RESULTS



Diameter of injector (d) = .002 m

$H/d=9$, $L/d=35$, $L_1/d=10$, $W/d=14$

| | Injected air | Main airstream |
|------------------------|--------------|----------------|
| Mach number | 1.0 | 2.06 |
| Static temperature (K) | 240.0 | 170.0 |
| Static pressure (KPa) | 152 | 36.0 |

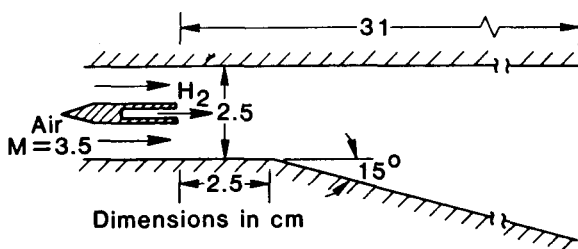


SHOCK TUNNEL SUPERSONIC COMBUSTION

At high hypersonic speeds, there are no steady state facilities capable of simulating the total enthalpy and pressure encountered in flight. The best experimental simulation of hypersonic combustion is conducted in pulse facilities that operate on the order of 1/1000 of a second. Results from such tests are shown in the figure for Mach 9 and 16 simulated flight conditions. Again CFD calculations were used to predict the flow. The Mach 16 results were not predicted as well as the lower speed test results. At these higher speeds where ground tests facilities are not available, CFD will have to be used to predict performance for NASP.

SHOCK TUNNEL SUPERSONIC COMBUSTION

University of Queensland

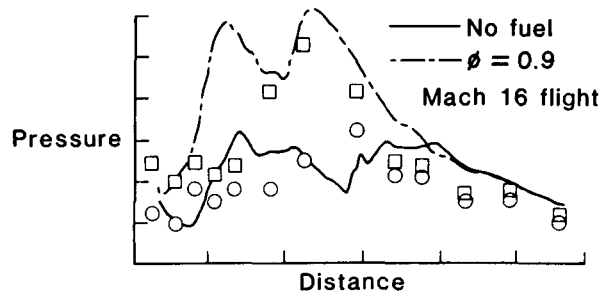
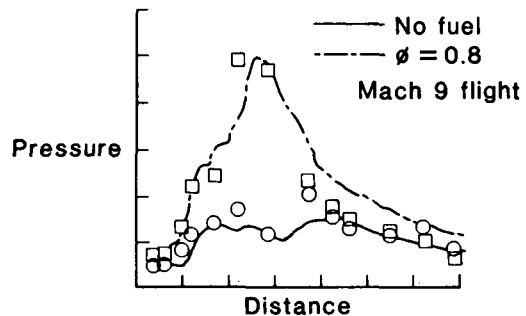


- Reasonable match between data and theory in Mach 8 to 12 speed range

- Indication of combustion effects at Mach 16

- Unexplained difference between data and theory
- Short residence time, low pressure, oxygen dissociation

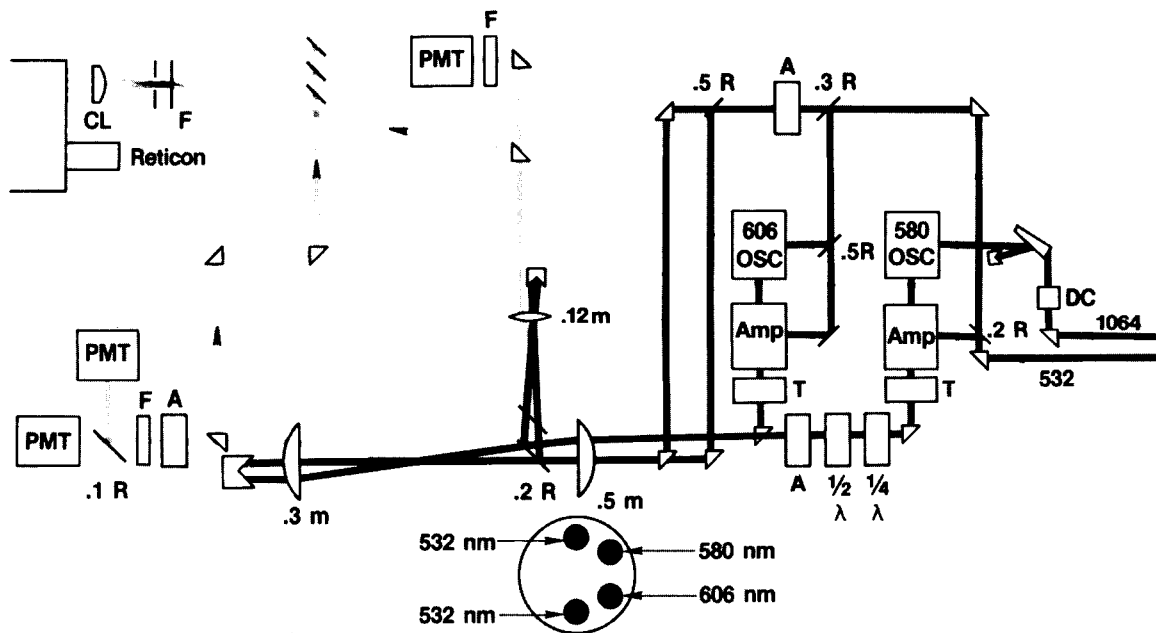
- Additional analysis and tests at higher pressure planned



MULTISPECIES CARS SCHEMATIC

In order to validate CFD codes, instream measurements of temperature and species profiles are required. In reacting supersonic streams it is difficult to use sampling probes for the measurements due to shock effects. Nonintrusive optical techniques are being developed to replace probe measurements. One technique being used is Coherent AntiStokes Raman Scattering, CARS. A schematic of the CARS system is shown. This system can measure temperature, oxygen, and nitrogen during a 10-nanosecond laser pulse at 10-Hz rate. The short pulse time allows measurements to be made in high speed turbulent flow.

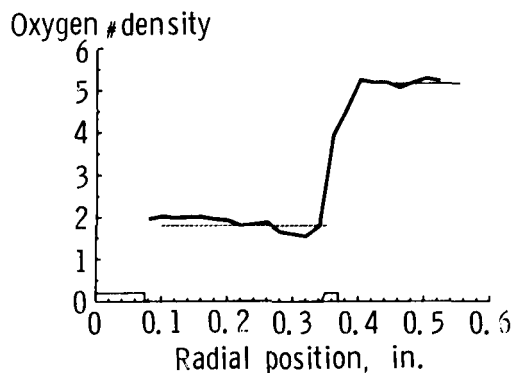
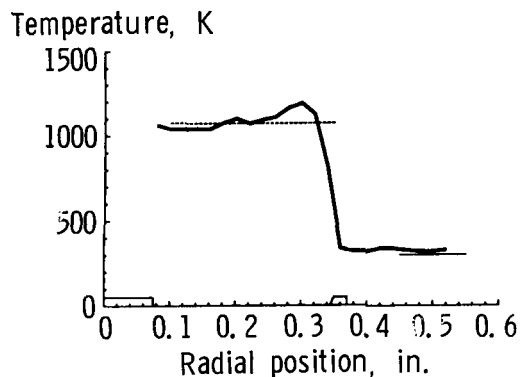
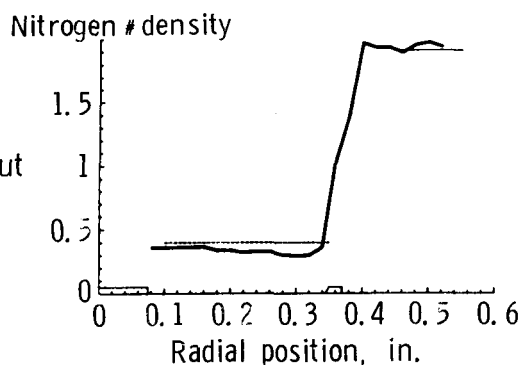
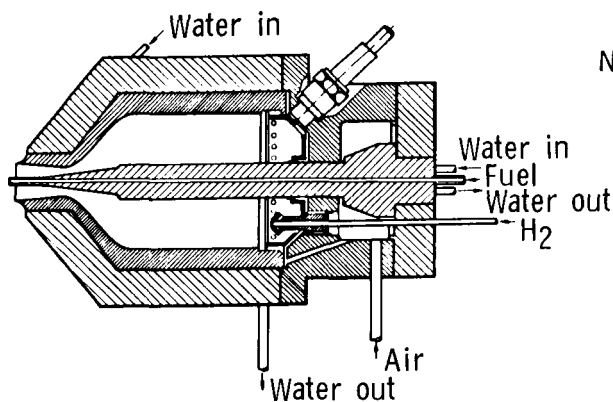
MULTISPECIES CARS SCHEMATIC



SUPERSONIC COMBUSTION EXPERIMENTS

A simple coaxial-jet, supersonic combustion experiment has been developed for CFD validation. The sonic jet is injected into a Mach 2 high enthalpy simulated air stream generated by the combustion and expansion of hydrogen and air with oxygen make up. A cross section of the apparatus is shown along with data taken in the exit plane. On the data plots the dashed lines are the calculated means and the solid lines are the CARS measurements. Tests are underway to make similar measurements to map the radial and axial profiles with and without fuel injection. This experiment is also being simulated by using a reacting flow CFD code.

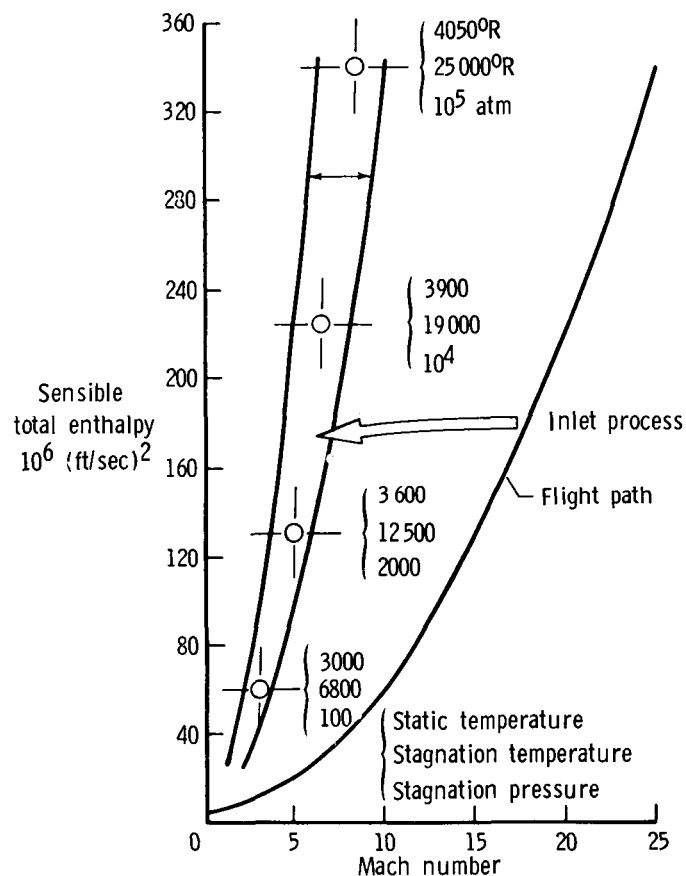
SUPERSONIC COMBUSTION EXPERIMENTS



SIMULATION REQUIREMENTS FOR HYPERSONIC SCRAMJET COMBUSTOR

In order to demonstrate scramjet performance in ground test, large high enthalpy facilities are required. This figure shows the total enthalpy required as a function of flight Mach number up to Mach 25 or near orbital speed. Also shown on the figure are the static temperatures, stagnation temperatures and the stagnation pressures required for proper simulation. The right most curve shows the flight path energy and the band shows the combustor flow conditions after the air is processed by the inlet. Even the combustor conditions rise rapidly with flight Mach number. In fact, hydrogen or air heaters can only be used to simulate flight speeds up to about Mach 8. Beyond this speed pulse facilities are required.

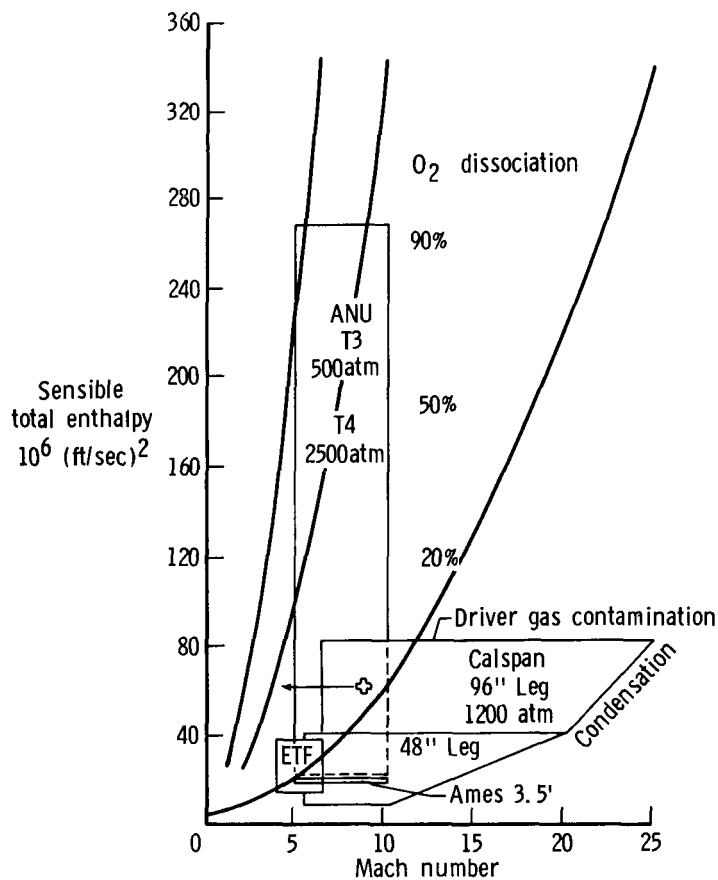
SIMULATION REQUIREMENTS FOR HYPERSONIC SCRAMJET COMBUSTOR TESTS



SIMULATION CAPABILITIES OF OPERATIONAL AND NEAR-TERM FACILITIES

This figure shows, on similar coordinates, the operational range of several combustion tunnels. Note that the T3 tunnel at the University of Queensland in Australia is the only facility that covers the higher Mach numbers. This facility has been used to produce flow simulating Mach 12.

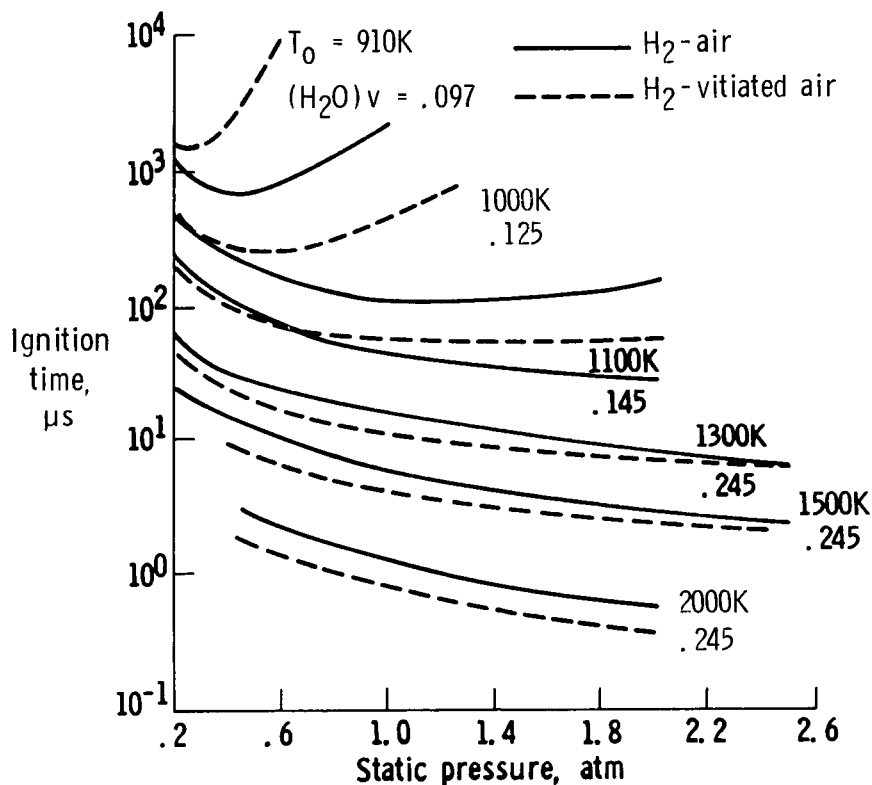
SIMULATION CAPABILITIES OF OPERATIONAL AND NEAR-TERM FACILITIES



EFFECTS OF VITIATION ON IGNITION TIME FOR STOICHIOMETRIC H_2 IN EQUILIBRIUM AIR

One of the problems associated with testing in high enthalpy ground test facilities is the fact that the energy is usually added to the flow by some mechanical, electrical or combustion process that contaminates or dissociates the air. Engine test facilities that use combustion of hydrogen or hydrocarbons can produce simulated Mach numbers of about 7 with oxygen content maintained at atmospheric levels. When these facilities are used, the tests are said to be conducted in vitiated air. This figure shows the effects of vitiation on the ignition delay of hydrogen as a function of static pressure and temperature. At low temperatures and pressures the effects become significant.

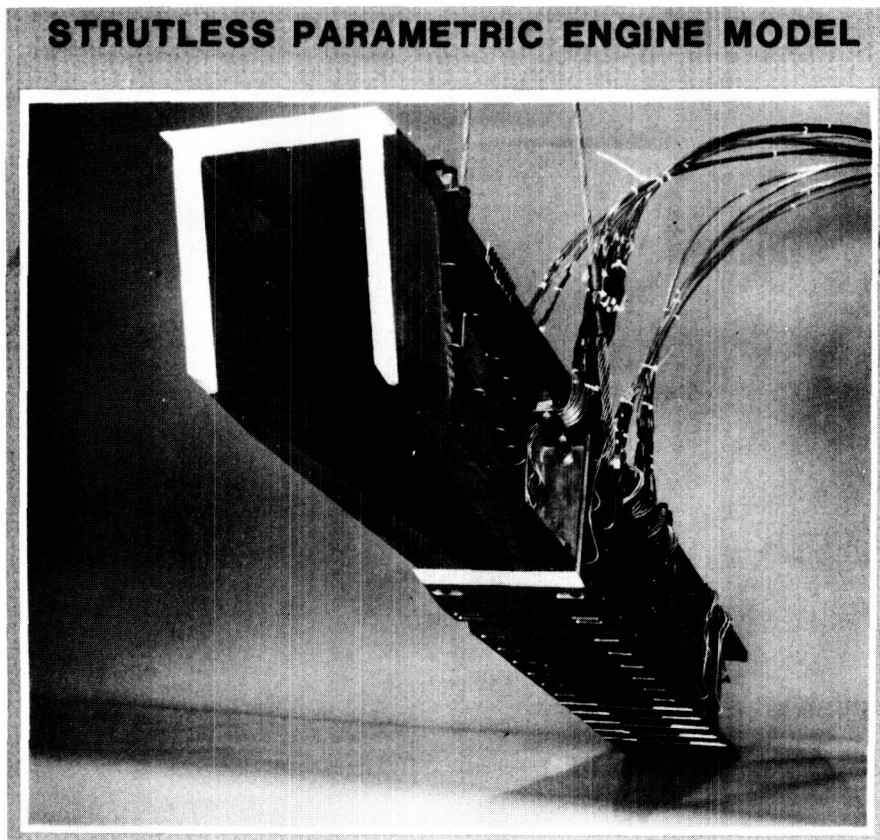
EFFECTS OF VITIATION ON IGNITION TIME FOR STOICHIOMETRIC H_2 IN EQUILIBRIUM AIR



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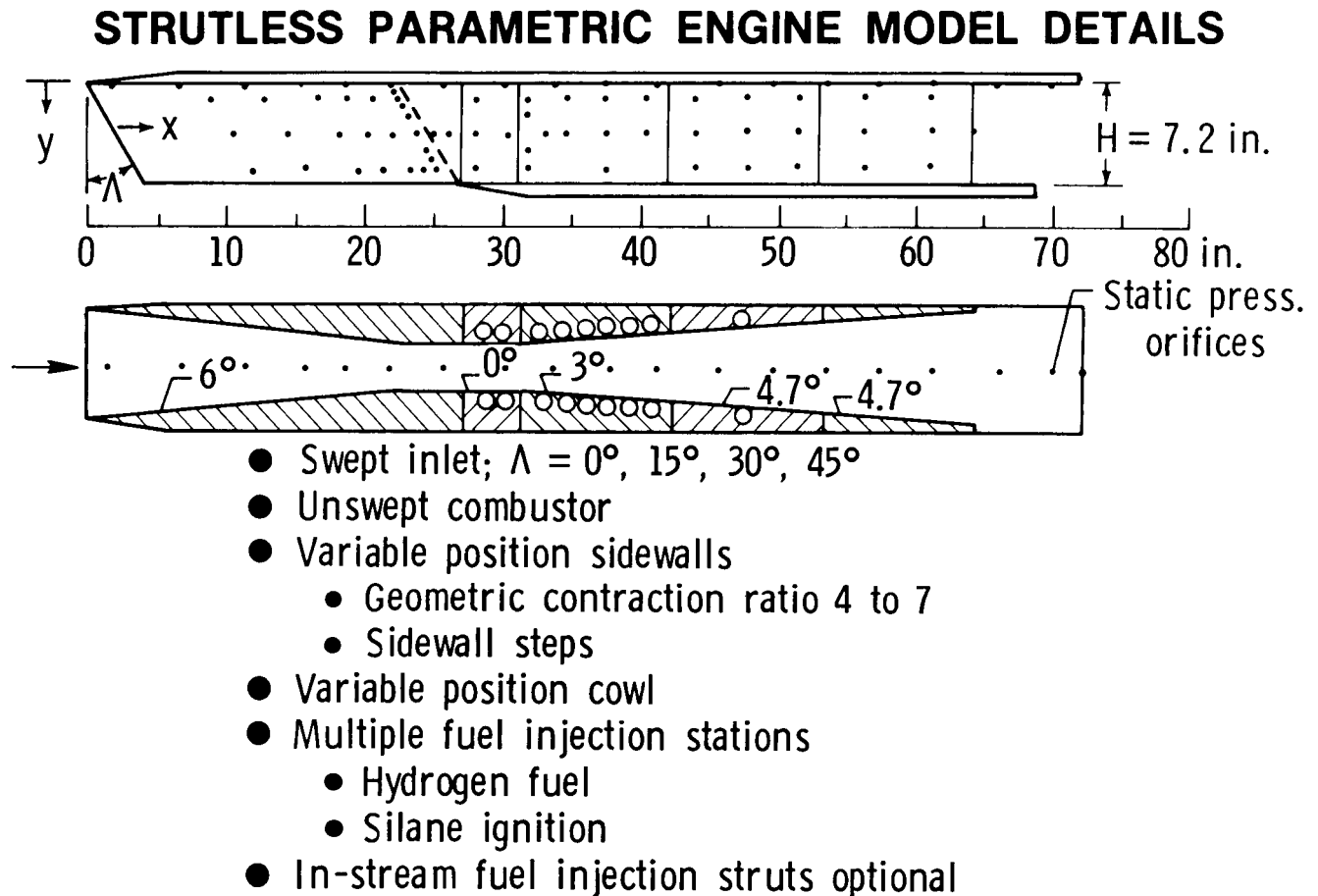
STRUTLESS PARAMETRIC ENGINE MODEL

In order to integrate the inlet and combustor technology, subscale engine tests are conducted in arc or vitiated scramjet test facilities at Langley from Mach 3.4 to 7 flight conditions. This figure shows the strutless parametric engine that is being used to study the sensitivity of performance to engine geometric variables, fuel injection location, fuel level, contraction ratio and inlet sweep.



STRUTLESS PARAMETRIC MODEL DETAILS

This sketch shows the details of the strutless model. The sidewalls are adjustable and the segmented side wall locations can be independently moved to tailor the internal geometry. Fuel can be injected from a number of side wall locations. The engine is mounted on a thrust balance in the tunnel and has numerous pressure ports to measure wall pressures.



MACH 4 ENGINE PERFORMANCE

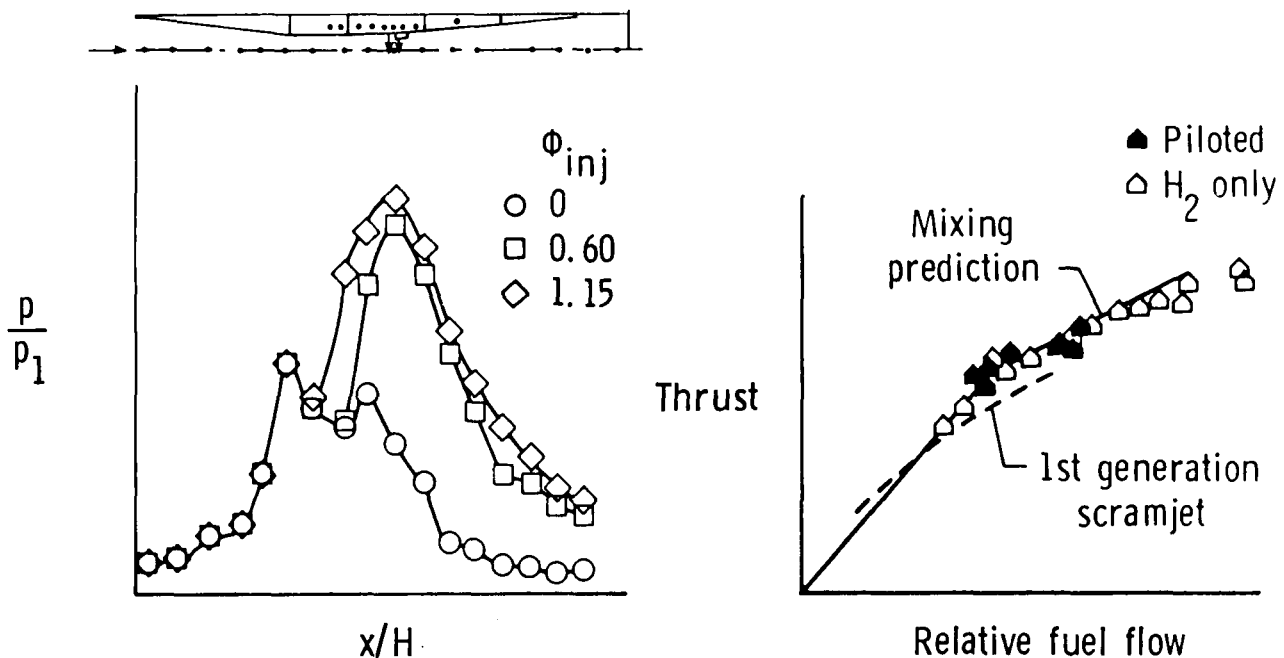
This figure shows typical pressure and force data obtained in Mach 4 tests of the strutless scramjet shown in the schematic. The peak pressure in the combustor increases as more fuel is injected into the engine and the pressure rise moves upstream closer to the inlet exit. However, no combustor-inlet interaction occurred since the combustion-induced pressure rise did not move onto the forward-facing sidewalls of the inlet.

The thrust curve shows a comparison of data from the strutless engine with that from an earlier version of a scramjet and with theory assuming mixing-controlled combustion. Solid symbols indicate that combustion of the primary hydrogen fuel was assisted by a pilot gas (silane/hydrogen) while the open symbols indicate that the fuel was all hydrogen. Agreement of the two sets of data (piloted and unpiloted) indicate that the wedges shown in the schematic were good flameholders. Agreement of the data with theory indicates that the combustion was mixing-controlled. And, finally, comparison of the data with that from the other scramjet shows the improvement in performance obtained by redesign.

MACH 4 ENGINE PERFORMANCE

STRUTLESS ENGINE

With wedges

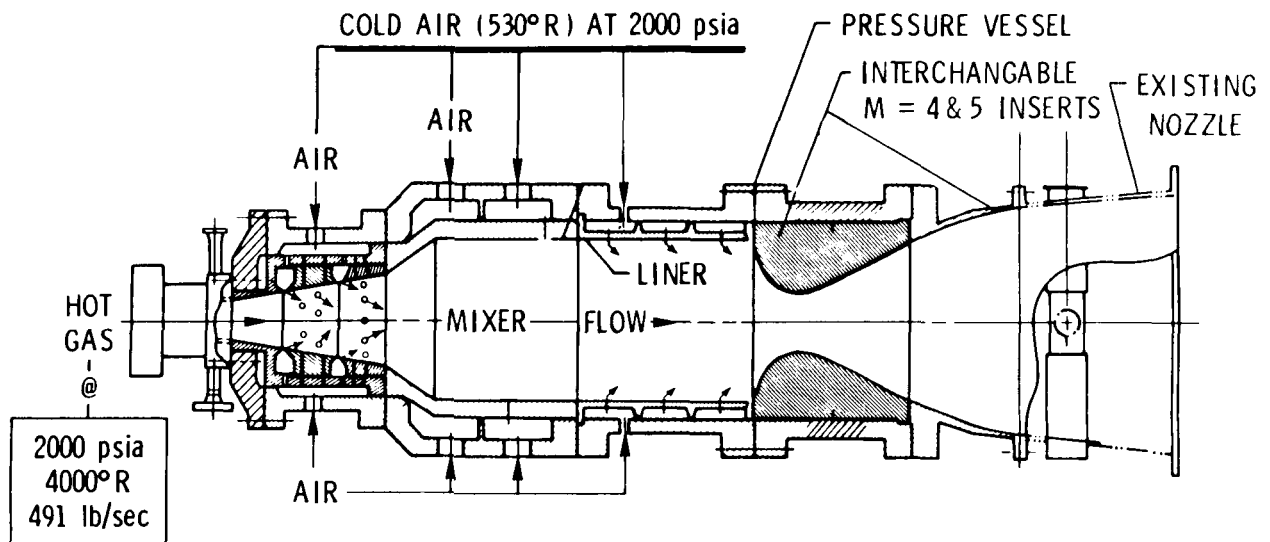


MIXER/ALTERNATE MACH NUMBER NOZZLES FOR 8-FOOT HTT

The Langley 8-foot test diameter high temperature tunnel is being modified for propulsion testing. When completed this facility will allow testing of larger scale models and complete missile size scramjets. This figure indicates the operating points at maximum total pressure of the 8-foot HTT with the Mach 4, 5, and 7 nozzles. A schematic is also shown of the mixer/throat section which is used with the Mach 4 and Mach 5 nozzles. When the Mach 7 nozzle is in service, the entire mixer/throat section (shown in section view) is replaced by the first portion of the Mach 7 nozzle. The hardware upstream and downstream of the cross-hatched section is used with all three nozzles.

During Mach 7 tests, no unheated air is added to the 4000 °R vitiated air exiting combustion heater. However, to obtain the lower Mach 4 and Mach 5 total enthalpies, unheated bypass air is added as shown in the schematic in a fashion similar to that employed in the Langley Arc-Heated Scramjet Test Facility.

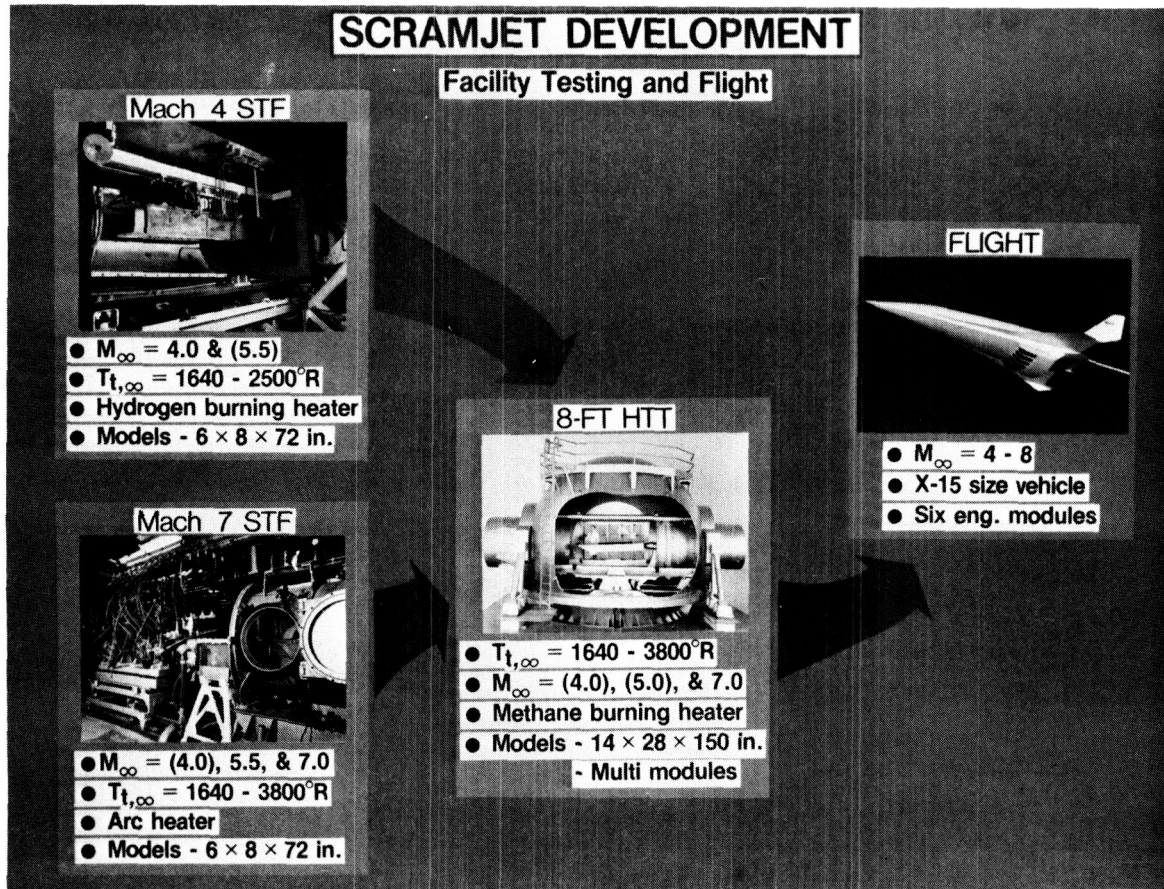
MIXER/ALTERNATE MACH NUMBER NOZZLES FOR 8-FOOT HTT



| MACH NUMBER | THROAT DIAMETER, in. | MIXER TOTAL PRESSURE, psia | MIXER TOTAL TEMPERATURE, °R | COLD AIR FLOW RATE, lb/sec | TOTAL FLOW RATE, lb/sec |
|-------------|----------------------|----------------------------|-----------------------------|----------------------------|-------------------------|
| 4.0 | 28.5 | 234 | 1640 | 1436 | 1927 |
| 5.0 | 17.1 | 444 | 2350 | 597 | 1088 |
| 7.0 | 5.6 | 2000 | 4000 | 0 | 491 |

SCRAMJET DEVELOPMENT

As high speed propulsion matures and new larger facilities become available, testing will move from single module subscale testing to testing of larger scale and multi-engine modules in the 8 ft tunnel and then to flight testing. The larger or near full scale tests will build confidence in the mixing and flameholding models and will allow realistic engine cooling evaluations in flight type hardware. The multi mode test will be used to study the influence of one module on the inlet flow of an adjacent module and allow the study of an inlet unstart on the total engine performance.





National Aeronautics and
Space Administration

Report Documentation Page

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|---|--|---|---|--|------------------------------|
| 1. Report No. NASA CP-10003 | | 2. Government Accession No. | | 3. Recipient's Catalog No. | |
| 4. Title and Subtitle Aeropropulsion '87 Session 6 - High-Speed Propulsion Technology | | | | 5. Report Date November 1987 | |
| | | | | 6. Performing Organization Code | |
| 7. Author(s) | | | | 8. Performing Organization Report No. E-3798 | |
| | | | | 10. Work Unit No. 505-62-3B | |
| 9. Performing Organization Name and Address National Aeronautics and Space Administration Lewis Research Center Cleveland, Ohio 44135-3191 | | | | 11. Contract or Grant No. | |
| | | | | 13. Type of Report and Period Covered Conference Publication | |
| 12. Sponsoring Agency Name and Address National Aeronautics and Space Administration Washington, D.C. 20546-0001 | | | | 14. Sponsoring Agency Code | |
| | | | | | |
| 15. Supplementary Notes Preprint includes figures and descriptive text. | | | | | |
| 16. Abstract NASA is conducting aeronautical research over a broad range of Mach numbers. In addition to the advanced CTOL propulsion research described in a separate session at this conference, the Lewis Research Center has intensified its efforts towards propulsion technology for selected high-speed flight applications. In a companion program, the Langley Research Center has also accomplished excellent research in Supersonic Combustion Ramjet (SCRAM) propulsion. What will be presented in this session is an unclassified review of some of the propulsion research results that are applicable for supersonic to hypersonic vehicles. This session not only provides a review of several key work areas, it also presents a viewpoint on future research directions by calling attention to the unique cycles, components, and facilities involved in this rapidly expanding field of work. | | | | | |
| 17. Key Words (Suggested by Author(s)) Aeronautical propulsion research; Materials; Structures; Internal fluid mechanics; Instrumentation; Controls; Subsonic propulsion; Supersonic propulsion; Hypersonic propulsion | | | 18. Distribution Statement Unclassified - Unlimited Subject Category 07 | | |
| 19. Security Classif. (of this report) Unclassified | | 20. Security Classif. (of this page) Unclassified | | 21. No of pages 118 | 22. Price* A06 |